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AERODYNAMIC CHARACTERISTICS OF A LARGE-SCALE UNSWEPT WING-BODY-TAIL CONFIGURATION WITH BLOWING APPLIED OVER THE FLAP AND WING LEADING EDGE

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SUMMARY

An investigation has been conducted in the Langley full-scale tunnel to determine the effects of a blowing boundary-layer-control lift-augmentation system on the aerodynamic characteristics of a large-scale model of a fighter-type airplane. The wing was unswept at the 70-percent-chord station, had an aspect ratio of 2.86, a taper ratio of 0.40, and 4-percent-thick biconvex airfoil sections parallel to the plane of symmetry. The tests were conducted over a range of angles of attack from approximately -4° to 23° for a Reynolds number of approximately 5.2×10^{6} which corresponds to a Mach number of 0.08. Blowing rates were normally restricted to values just sufficient to control air-flow separation.

The results of this investigation showed that wing leading-edge blowing in combination with large values of wing leading-edge-flap deflection was a very effective leading-edge flow-control device for wings having highly loaded trailing-edge flaps. With leading-edge blowing there was no hysteresis of the lift, drag, and pitching-moment characteristics upon recovery from stall. End plates were found to improve the lift and drag characteristics of the test configuration in the moderate angle-of-attack range, and blockage to one-quarter of the blowing-slot area was not detrimental to the aerodynamic characteristics. Blowing boundary-layer control resulted in a considerably reduced landing speed and reduced landing and take-off distances. The ailerons were very effective lateral-control devices when used with blowing flaps.

INTRODUCTION

The use of thin, low-aspect-ratio, unswept and sweptback wings on modern fighter-type airplanes has seriously limited the low-speed maximum lift and reduced the longitudinal stability and has, therefore, seriously limited the low-speed performance of these aircraft. Recent

wind-tunnel investigations of boundary-layer control by blowing over the wing leading- and trailing-edge flaps of highly sweptback-wing configurations (refs. 1 to 3) have shown significant improvements in the maximum lift and longitudinal stability characteristics of these configurations. To date, however, the only systematic boundary-layer-control work that has been done toward improving the low-speed aerodynamic characteristics of high-speed configurations with thin, unswept wings was that reported in reference 4.

Because of the limited amount of information available about configurations of this type, an investigation has been conducted in the Langley full-scale tunnel to determine the effect of a blowing boundary-layer-control lift-augmentation system on the low-speed aerodynamic characteristics of a large-scale model of a fighter-type airplane. The wing was unswept at the 70-percent-chord station, had an aspect ratio of 2.86, a taper ratio of 0.40, and 4-percent-thick biconvex airfoil sections parallel to the plane of symmetry. The horizontal tail was unswept at the 50-percent-chord station, had an aspect ratio of 3.33, a taper ratio of 0.50, and 4-percent-thick airfoil sections parallel to the plane of symmetry.

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For the present investigation, emphasis was placed on increasing maximum lift while maintaining longitudinal stability to maximum lift, determining the most desirable horizontal-tail height for longitudinal stability and control, determining a lateral-control device suitable for use with a high-lift blowing boundary-layer-control system, and estimating the effects of wing leading- and trailing-edge blowing on the low-speed landing and take-off performance characteristics.

The investigation was conducted for a range of angles of attack from approximately -4° to 23° for a Reynolds number of approximately 5.2×10^{6} which corresponds to a Mach number of 0.08.

SYMBOLS AND COEFFICIENTS

b	wing span, ft
c	local wing chord, ft
c_{av}	average wing chord S/b, ft
$\bar{c}$	wing mean aerodynamic chord $\frac{2}{S}\int_0^{b/2} c^2 dy$, ft

$\bar{c}_t$ horizontal-tail mean aerodynamic chord, ft

G weight rate of air ejected from blowing slot, lb/sec

g acceleration due to gravity, ft/sec^2

h_d deflector projection, ft

h_s spoiler projection, ft

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i_t incidence of horizontal tail, trailing edge down, positive, deg

l fuselage length, ft

p local static pressure, lb/sq ft

p_∞ free-stream static pressure, lb/sq ft

Q volume rate of air ejected from blowing slot, cu ft/sec

q_∞ free-stream dynamic pressure, lb/sq ft

r fuselage radius at any longitudinal station, ft

S wing area, sq ft

S_t horizontal-tail area, sq ft

V airplane configuration flight speed, ft/sec

V_j velocity of ejected air at slot, ft/sec

V_∞ free-stream velocity, ft/sec

x chordwise distance measured parallel to the plane of symmetry, ft

y lateral distance measured perpendicular to the vertical plane of symmetry, ft

$\dot{Z}$ vertical velocity of airplane configuration, ft/sec

z vertical height of horizontal tail measured from fuselage center line (above center line, positive), ft

α	angle of attack, deg
γ	glide-path or climb angle of airplane configuration, deg
δ	deflection, perpendicular to hinge line, of the leading- and trailing-edge flaps and ailerons, deg
ρ_∞	mass density of free-stream air, slugs/cu ft
C_D	drag coefficient, $\dfrac{\text{Drag}}{q_\infty S}$
C_L	lift coefficient, $\dfrac{\text{Lift}}{q_\infty S}$
ΔC_L	increment of lift coefficient
C_l	rolling-moment coefficient, $\dfrac{\text{Rolling moment}}{q_\infty Sb}$
ΔC_l	increment of rolling-moment coefficient
C_m	pitching-moment coefficient, $\dfrac{\text{Pitching moment}}{q_\infty S\bar{c}}$
C_N	wing normal-force coefficient, $\displaystyle\int_0^{1.0} c_n \frac{c}{c_{av}}\, d\left(\frac{y}{b/2}\right)$
C_n	yawing-moment coefficient, $\dfrac{\text{Yawing moment}}{qSb}$
ΔC_n	increment of yawing-moment coefficient
C_p	pressure coefficient, $\dfrac{p - p_\infty}{q_\infty}$
C_μ	blowing jet momentum coefficient, $\dfrac{GV_j}{gq_\infty S}$
$\dfrac{dC_L}{dC_\mu}$	rate of change of lift coefficient with blowing jet momentum coefficient
c_n	wing section normal-force coefficient, $\displaystyle\int_0^{1.0} C_p\, d\left(\frac{x}{c}\right)$

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Subscripts:

a aileron (use of subscript "a" without further subscript "R" or "L" indicates both ailerons drooped, trailing edge down, positive)

f trailing-edge flap (trailing edge down, positive)

k knee of wing leading-edge flap

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n wing leading-edge flap (leading edge down, positive)

L left hand

R right hand

T denotes total aileron deflection

MODEL

The geometric characteristics for the large-scale model used in this investigation are shown in figure 1. The wing was unswept at the 70-percent-chord station, had an aspect ratio of 2.86, a taper ratio of 0.40, and 4-percent-thick biconvex airfoil sections parallel to the plane of symmetry. The horizontal tail was unswept at the 50-percent-chord station, had an aspect ratio of 3.33, a taper ratio of 0.50, and 4-percent-thick biconvex airfoil sections parallel to the plane of symmetry.

Photographs of the model mounted for tests in the Langley full-scale tunnel are given as figure 2. Details of the flow-control devices on the wing are given in figure 3.

The wing was equipped with 30-percent-chord flaps and ailerons (measured from the hinge line) with the ailerons being capable of deflection as outboard flaps. The spanwise lengths of the flaps and ailerons were 0.55b/2 and 0.30b/2, respectively. For convenience, the 0.55b/2 flap will be referred to as the "half-span" flap, and the flap-aileron combination, when used as a flap, will be referred to as the "full-span" flap. The flaps and ailerons had a full-length, 0.010-inch-gap blowing slot located in the nose radius (figs. 3(a) and 3(c)) which became exposed at a deflection angle of about 40°.

The wing leading-edge flow-control device was a 15-percent-chord, full-span, leading-edge flap with a full-length, 0.010-inch-gap blowing

slot located at the knee of the flap. (See fig. 3(b).) The blowing slot became exposed at a flap-deflection angle of about 20°.

The wing was also equipped with spoilers and deflectors on the left-hand wing panel. The spanwise extent of these devices is shown in figure 1 with a detailed drawing shown in figure 3(c) and a general view shown in figures 4(a) and 4(b). The various segments of the spoilers and deflectors are referred to as 1, 2, 3, and 4 as shown in figure 1. The device referred to as 5 consists of the 25- to 50-percent span of device 3. When the spoiler and deflector were deflected simultaneously, a slot was formed through the wing making what is generally called a spoiler-slot-deflector configuration. In this paper this configuration will be referred to as a spoiler-deflector configuration. For all configurations in which the spoiler-deflector combination was used, the ratio of spoiler-to-deflector projection was 2 to 1.

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The horizontal tail was all movable, could be mounted at three tail heights ($z/\bar{c}$ of -0.09, 0.40, and 0.80), and was located at a tail length of $1.87\bar{c}$.

All of the blowing-slot ducts (figs. 3(a) and (b)) were tapered toward the wing tips so that a uniform slot pressure was obtained over the full length of the slot. The wing leading-edge-flap, aileron, and trailing-edge-flap duct pressures were individually controlled to provide for regulation of the boundary-layer-control air flow.

The wing was equipped with end plates for two test conditions. Photographs of these end plates, along with photographs of spoiler 1 and a portion of deflector 1, are given as figure 4. The end plates, mounted symmetrically at the wing tips, were 6 feet long and 2 feet high with rounded corners of 1-foot radius.

Chordwise surface pressure orifices were located on the upper and lower surfaces of the left-hand wing panel and on the left-hand one-half of the fuselage. The spanwise orifice stations, hereinafter referred to as stations 1 to 7 as indicated in figure 5, were referenced from the fuselage center line and were 0, 15.4, 22.1, 42.6, 64.0, 80.0, and 91.8 percent of the semispan, respectively. Station 2 was actually located on the fuselage surface 60° from the vertical plane of symmetry. The value for $\frac{y}{b/2}$ of 0.154 was arbitrarily chosen for plotting purposes to be an average value. The location of the fuselage orifices and the coordinates of the fuselage are given in figure 5.

AIR SUPPLY

The air used for boundary-layer control was supplied by a compressor capable of delivering to the model, at full flow and at a pressure ratio of 3.0, a maximum of 12 pounds of air per second (maximum for present tests was about 2 pounds per second). The compressor was isolated from the model, and air was delivered through a system of ducting. The air was brought onto the scale-balance-frame supporting the model by flexible connectors alined so that reaction forces would
L cancel. In order to permit angle-of-attack change, an air-tight slip-
9 joint was located on the lateral axis of rotation between the fuselage
2 plenum and the air-supply pipe entering the model through the bottom
7 of the fuselage.

TESTS

The static longitudinal stability and control and the lateral control characteristics of the model were determined from force measurements obtained from the tunnel scale-balance system for a range of angles of attack from approximately -4° to 23°. Surface-pressure-distribution data were obtained for several of the test configurations to show the air-load distribution over the individual and collective parts of the model.

Preliminary tests showed that woolen tufts attached to the upper surface of the wing and horizontal tail had negligible effects on the force and moment characteristics and pressure coefficients of the model and, therefore, were left installed for flow-visualization studies throughout the investigation.

All of the tests were conducted for a Reynolds number of about 5.2×10^6 which corresponds to a Mach number of 0.08. An index of the test conditions for the various configurations used in the investigation is given in the following table:

δ_n, deg	δ_f, deg	$\delta_{a,L}$, deg	$\delta_{a,R}$, deg	Spoiler	Spoiler-deflector	i_t, deg	Tail height, $z/\bar{c}$	$C_{\mu,k}$	$C_{\mu,f}$	$C_{\mu,a}$	Remarks	
0*, 10, 15, 20, 30	0	0	0	---------	---------	0	-0.09	0	0	0	Wing leading-edge and trailing-edge flap effectiveness	No boundary-layer control
0	30, 37, 47	0	0	---------	---------	0	-0.09	0	0	0		
	30, 37, 47	30, 37, 47	30, 37, 47									
20	30, 37, 47	0	0	---------	---------	0	-0.09	0	0	0		
	30, 37, 47	30, 37, 47	30, 37, 47									
30	30, 37*, 47	0	0	---------	---------	0	-0.09	0	0	0		
	37*, 30, 47	37, 30, 47	37, 30, 47									
40	30, 37, 47	0	0	---------	---------	0	-0.09	0	0	0		
	30, 37, 47	30, 37, 47	30, 37, 47									
50	37, 47	0	0	---------	---------	0	-0.09	0	0	0		
	37, 47	37, 47	37, 47									
40	37, 47	0	0	---------	---------	0	-0.09	0	0.012	0, 0.004		
	37, 47	37, 47	37, 47									
50	37, 47*, 60	0	0	---------	---------	0	-0.09	0	0.012	0, 0.004		
	37, 47, 60	37, 47, 60	37, 47, 60									
40	37*, 47	0	0	---------	---------	0	-0.09	0.010	0.012	0, 0.004		Trailing-edge blowing only
	37*, 47	37, 47	37, 47									
50	37, 47*	0	0	---------	---------	0	-0.09	0.010	0.012	0, 0.004		
	37*, 47*, 60*	37, 47, 60	37, 47, 60									
0	0	0	0	---------	---------	-17.9 to 10	-0.09, 0.40, 0.80	0	0	0	Tail height and tail effect	Leading- and trailing-edge blowing
30	37	0	0	---------	---------	-17.9 to 10	-0.09, 0.40, 0.80	0	0	0		
	37	37	37									
50	47	0	0	---------	---------	-17.9 to 10	-0.09, 0.40, 0.80	0.010	0.012	0, 0.004		
	47	47	47									
50*	47	-14 to 37	0	---------	---------	0	-0.09	0.010	0.012	0	Aileron effect	
	47	18 to 60	37	---------	---------	0	-0.09	0.010	0.012	0.004		
50	47	0	0	2-3, 3	---------	0	-0.09	0.010	0.012	0	Spoiler effect	
	47*	47	47	1-2, 2-3, 3-4, 3, 5	---------	0	-0.09	0.010	0.012	0.004		
	47	0	0	2-3, 3	2-3, 3	0	-0.09	0.010	0.012	0	Spoiler-deflector effect	
	47*	47	47	1-2, 2-3, 3-4, 3	1-2, 2-3, 3-4, 3	0	-0.09	0.010	0.012	0.004		
50*	47	47	47	---------	---------	0	-0.09	0.010, 0.019	0.012, 0.019	0.004, 0.007	End plates	
50	47	47	47	---------	---------	0	-0.09	0.011	0.014	0.005	Slot blockage	

*Indicates pressure-distribution data presented as well as normal scale-balance force and moment data.

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METHODS AND CORRECTIONS

The mass flow of air being ejected from the individual blowing slots was calculated from measurements of the individual duct pressure, temperature, and slot-exit area. Several shielded total-pressure tubes were located within each duct to ascertain that uniform flow was achieved along the length of the slot. Duct pressures were indicated on a mercury manometer and slot areas were measured with test pressure applied.

The surface static pressures, measured on a multiple-tube manometer and photographically recorded, were reduced to coefficient form by electronic step-integration processes. With trailing-edge-flap blowing applied, the flap-chord forces were included in the appropriate calculations. For tests without trailing-edge-flap blowing, the flap-chord forces were found to be negligible and were not included in the calculations.

The determination of the fuselage loading and the summation of this loading and the wing loading to obtain the total force coefficients required considerable manipulation of the fuselage-pressure data. The method used for calculating the fuselage loading is given in appendix A.

The force and moment data as obtained from the tunnel scale system have been corrected for airstream misalinement, buoyancy, and jet-boundary effects. In order to make the data equivalent to a self-contained system, the drag coefficients were corrected by adding to the drag the term $\rho_\infty QV$, which is the drag equivalent of taking on board a mass of air $\rho_\infty Q$ having an original velocity relative to the model of V. This correction was necessary because the air ejected from the model was admitted from a source that had a zero component of momentum in the free-stream direction. The force and moment data, as presented, contain the effect of jet momentum because this would be reflected in the aerodynamic characteristics of an airplane with boundary-layer-control devices.

The pressure-distribution data were corrected for the average effects of airstream misalinement and jet-boundary effects on the angle of attack.

RESULTS AND DISCUSSION

Longitudinal Characteristics

Basic data for configurations without boundary-layer control.- The results of the tests without boundary-layer control are shown in

figures 6 and 7. These tests were conducted for the low tail position of $z/\bar{c} = -0.09$. The basic configuration (without flaps deflected) had a maximum lift coefficient of about 0.8, and the configuration was longitudinally stable throughout the lift range. In all cases, full-span trailing-edge flaps produced higher values of lift coefficient than did the comparable half-span flap configuration, and leading-edge-flap deflection was very beneficial for either trailing-edge-flap configuration. The wing leading-edge flaps reduced lift at low angles of attack; however, the maximum lift and the angle of attack at which it occurred were greatly increased when wing leading-edge flaps were added to the trailing-edge-flap configurations because of delayed wing-leading-edge L
air-flow separation. It can be readily seen, however, that a limit 9
exists for increasing maximum lift by leading-edge-flap deflection since 2
increasing the leading-edge-flap deflection from 40° to 50° resulted in 7
a large loss in maximum lift.

The configurations producing the greatest maximum lift were with half- or full-span trailing-edge flaps deflected 47° and wing leading-edge flaps deflected 40°. (See figs. 6(e) and 6(f).) These configurations were also longitudinally stable or neutrally stable through the lift range. Because the configurations with half- or full-span trailing-edge flaps deflected 37° and leading-edge flaps deflected 30° appeared to be the best compromise between maximum lift and good longitudinal stability through the lift range, these configurations were selected arbitrarily for comparison with configurations with boundary-layer control to be presented subsequently.

A few tests were conducted with only the wing leading-edge flap deflected, and the results of these tests are shown in figure 7. The drag was appreciably reduced for lift coefficients greater than about 0.3; however, at angle of attack of 0° the configuration without flaps deflected had the lowest drag.

Basic data for configurations with boundary-layer control.- At the beginning of the boundary-layer-control tests it was desirable to establish the minimum blowing boundary-layer-control requirements for the prevention of air-flow separation over the trailing-edge flaps at an angle of attack of 0°. For this angle of attack, very little air-flow separation existed forward of the flaps; so the flap blowing requirements should be fairly accurately defined. Wing leading-edge blowing over a highly deflected leading-edge flap was to be used for air-flow control over the wing forward of the flap at angles of attack. Because the leading-edge blowing would eliminate the air-flow separation forward of the flap, the flap blowing requirement at angles of attack should be essentially the same as that established at $\alpha = 0^{\circ}$. Several tests at $\alpha = 0^{\circ}$ were therefore conducted for both half- and full-span trailing-edge-flap configurations for values of trailing-edge-flap blowing momentum coefficient varying from 0 to about 0.018. The results of these

tests are shown in figure 8. For half-span flaps the blowing coefficient C_μ required was only 0.004 to 0.005 for flap deflections of 37° and 47°, respectively. For the full-span flap, the value of C_μ required to prevent air-flow separation was about 0.003 for the flap deflected 37° but was about 0.012 for the flap deflected 47°. In order to insure that sufficient blowing rates were used for the remainder of the tests, a value for C_μ of 0.012 was selected for use with the half-span flap and a value of 0.016 for the full-span flap (0.004 for the aileron).

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The rate of change of lift coefficient with blowing-jet momentum coefficient dC_L/dC_μ shown by the dashed line in figure 8 was utilized in the landing performance calculations described in appendix B.

Effect of high-lift and flow-control devices in combination with boundary-layer control.- The effects of trailing-edge-flap blowing and wing leading-edge deflection and blowing on the aerodynamic characteristics of several half- and full-span trailing-edge-flap configurations are shown in figure 9. For comparison purposes some of the data without boundary-layer control are repeated.

Many previous investigations of flap blowing configurations on swept wings have shown that highly loaded trailing-edge flaps without some form of wing leading-edge flow-control device provided a large increase in lift at low to moderate angles of attack but provided no increase in $C_{L,max}$ over that obtained for configurations without boundary-layer control. It was assumed that this same variation of $C_{L,max}$ with leading-edge device would occur in the present case, so the determination of the effects of flap blowing alone was not included in the present investigation. All of the flap blowing tests were conducted with the wing leading-edge flap deflected.

By using the assumption that trailing-edge-flap blowing alone does not provide an increase in $C_{L,max}$, leading-edge-flap deflection to 40° is seen (fig. 9) to provide a large increase in $C_{L,max}$ for flap blowing configurations. Deflecting the leading-edge flap more than 40°, however, is seen to result in a large loss in lift of the half-span flap blowing configurations. The observation of woolen tufts attached to the wing surface showed that separation was occurring at the knee of the 50° drooped leading edge, and this separation was in turn detrimental to the loading of the trailing-edge flap. It was reasoned, therefore, that the application of blowing at the knee of the drooped leading edge would at least delay this separation to higher angles of attack and result in higher values of $C_{L,max}$.

The addition of wing leading-edge blowing at the knee of both the 40° and 50° drooped leading edge is seen (fig. 9) to increase the maximum

lift of all configurations. For the configurations already having relatively high values of $C_{L,max}$, the leading-edge blowing only increased the value of $C_{L,max}$ by about 0.1; however, for the half-span-flap configurations having the large $C_{L,max}$ loss with leading-edge droop to 50°, the loss in lift was eliminated and these configurations produced values of $C_{L,max}$ greater than any of the other half-span flap configurations.

Several cursory hysteresis data points were taken for various configurations while angle of attack was decreased from values greater than the stall angle to values somewhat lower. It was found that configurations with wing leading-edge blowing had very little or no hysteresis of the lift, drag, and pitching-moment data. In the event of stall, the airplane would recover its unstalled characteristics as soon as the angle of attack was reduced below the angle of stall.

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Slot blockage becomes a problem on a production aircraft because a long, uninterrupted slot would be very difficult to build, and during flight the flexibility of a wing would probably close the slot in some places and open it more in others. Spacers (blockage) would probably be required to maintain the slot gap. Tests were therefore conducted with the various blowing slots partially blocked. When one-quarter of the area of the slot was blocked (1/2 inch of length blocked and $1\frac{1}{2}$ inches open) and the value of C_μ was approximately the same as that used for tests with the slot open, no detrimental effect on the aerodynamic characteristics was noted. (See fig. 10.) The slight increase in lift noted for the configuration with the partially blocked slot was believed to be caused by the slight increase in the value of C_μ. When the blockage was increased to one-half the slot area, however, woolen tufts attached to the wing surface showed the air flow over the surface to be very poor and the force test was discontinued. It was surmised that an appreciable loss in lift in the moderate to high angle-of-attack range would have resulted from blockage of one-half the slot area.

In order to determine the general effect on the lift, drag, and pitching moments of installing wing-tip tanks, outboard engines, or some similar device, end plates were installed at the wing tips of the configuration with full-span trailing-edge flaps and leading- and trailing-edge blowing. Photographs of the end-plate installation are given as figures 4(c) and 4(d). The results of the end-plate tests along with results obtained when the blowing rate was arbitrarily increased about 70 percent with end plates installed are given in figure 10. The end plates increased the lift coefficient by about 0.10 in the low angle-of-attack range and by about 0.15 in the moderate to high range. The maximum lift coefficient, however, was improved only

about 0.05. Increasing the blowing rate by about 70 percent provided a similar but larger increase in lift in the low to moderate angle-of-attack range but increased the maximum lift coefficient by about 0.25 $\left(C_{L,max} = 2.5\right)$. The use of the end plates caused an appreciable reduction in drag for a given lift coefficient with the amount of this decrease becoming greater with increasing lift. End plates or some device acting as an end plate, therefore, might well be used with beneficial results on an airplane similar to the present test configuration.

Effect of horizontal-tail height.- The low tail position $(z/\bar{c} = -0.09)$ was used for all the previously presented data because from many previous blowing boundary-layer-control investigations the low tail position seemed to provide the best longitudinal stability characteristics. By utilizing this one tail position, the boundary-layer-control requirements were also quickly ascertained. In order to determine whether the low tail position was indeed a better position than a somewhat higher position, several tests were conducted with and without boundary-layer control applied for tail heights $z/\bar{c}$ of -0.09, 0.40, and 0.80 and with the horizontal tail off. (See fig. 11.) The low tail position appeared to provide the best longitudinal stability characteristics for all configurations except the one with half-span flaps deflected with blowing applied. This configuration appeared to be slightly better with the tail in the middle position.

Effect of horizontal-tail deflection.- Although the low tail position in general resulted in better longitudinal stability characteristics than did the middle and high tail at 0° tail incidence, the low tail was not sufficiently better than the other positions to preclude their use. Horizontal-tail-incidence tests were therefore conducted for several configurations for all three tail heights. Results of these tests are shown in figures 12 to 14.

The horizontal tail is unable to trim the high-lift configurations to maximum lift without producing a neutrally stable or an unstable configuration. This instability, however, does not preclude the use of the high-lift devices because, upon examination of the data, the instability is seen to be the result of horizontal-tail stall. Even at zero incidence at low angles of attack, the horizontal tail is stalled for some configurations. In order to trim an airplane of this type, a high-lift horizontal tail would be required. For the tail length and geometry of the present tail, the maximum tail lift coefficient would be about 0.8 (basic wing data) which would produce an increment of pitching-moment coefficient of about 0.3. This increment obviously would not be sufficient to trim the model in many cases. The problem of trim, therefore, is resolved (in the present case) into a problem of increasing the lift on the tail. This increase could be accomplished by several means with or without boundary-layer control on the tail. For instance, adding leading- and trailing-edge flaps to the horizontal tail would almost

double the maximum lift coefficient of the tail which would provide sufficient trim for all of the configurations presented.

Lateral Control Characteristics

With the longitudinal characteristics fairly well defined as acceptable for some configurations, it was desirable to determine the lateral control characteristics. Inasmuch as innumerable lateral-control investigations have been conducted for configurations which did not have boundary-layer control, only the lateral control characteristics for configurations with boundary-layer control will be discussed herein.

Effect of aileron deflection.- The lift, rolling-moment, and yawing-moment coefficients resulting from deflection of the left-hand aileron of the half- and full-span flap configurations are shown in figure 15. These data are reduced to incremental values in figure 16 by assuming a neutral aileron position and from this point combining the incremental force or moment coefficients resulting from an up and down deflection of the left-hand aileron. For the half-span flap the neutral position is assumed to be the nondeflected position, and for the full-span flap the aileron neutral position is assumed to be deflected downward to 30°. The data of figure 16(b) were actually taken from data obtained with the right-hand aileron base condition at a deflection of 47° (fig. 15(b)). This is believed to be unimportant, however, in that only the incremental values obtained from left-hand aileron deflection are to be discussed. The up-to-down deflection ratio of the ailerons was taken as 1 to 2.

The aileron control characteristics of both the half- and full-span-flap configurations are shown in figure 16. These data show that the ailerons produce an almost linear variation of rolling-moment coefficient with deflection with sufficient roll power to produce the desired rate of roll, at moderate to high angles of attack, for a configuration of this type. A value of C_l of about 0.04 is all that is required for a value of $pb/2V$ of about 0.09 - the value normally used for a fighter-type airplane.

Deflection of the left- and right-hand ailerons would result in a negligible overall change in lift. The adverse yawing moments produced by the aileron deflection were small for the half-span-flap configuration. The yawing moments produced by aileron deflection on the full-span-flap configuration were considerably larger than for the half-span flap case; however, a normal rudder installation could easily control these moments. Ailerons on a blowing boundary-layer-control configuration of the subject type, therefore, would be a very good low-speed lateral control device.

It should be noted that the initial rolling-moment coefficient shown for the base conditions (see figs. 15(a) and 15(b)) is assumed to be a combination of asymmetry in the model construction and high-lift and flow-control devices installation and deflection, and to some extent - asymmetric blowing. It is believed to be unimportant for these data, however, in that the incremental values are used for the discussion, and the woolen tufts attached to the wing surface did not show any large differences in the air flow over the left- and right-hand wing panels.

Effect of spoiler and spoiler-deflector deflection.- Another lateral-control device which has received much research attention, especially at high speed, has been the spoiler and the spoiler-deflector combination. In order to determine the low-speed lateral control characteristics of these types of devices when used in combination with blowing boundary-layer control, several tests were conducted utilizing several combinations of spoilers and spoiler deflectors. Results of these tests on a full-span flap configuration are shown in figures 17 and 18. These data, reduced to incremental values, are shown in figure 19.

Both spoiler and spoiler-deflector combinations were very powerful roll-producing devices; however, the variation of rolling moment with projection was very nonlinear, and the required amount of rolling moment produced by the control was obtained with very small spoiler or spoiler-deflector projections. Even the small spanwise segment of control, referred to as number 5, produced the required amount of roll with a very small projection. The reason this small segment of spoiler was so effective is believed to be because of its unique position of being at a spanwise station that is extremely sensitive to a disturbance of any kind. The segment is forward of the most heavily loaded portion of the flap, and the disturbance created by its projection could be expected to produce a large loss in lift and therefore result in a large rolling moment.

A few tests were conducted with the spoilers and spoiler-deflector combinations on a half-span blowing-flap configuration. The results of these tests are shown in figure 20. The spoiler effectiveness, as indicated in the present case by the shape of the curve of ΔC_l plotted against percent projection, was very poor in the low projection range (0 to about 1.5 percent) after which there was a range of high effectiveness followed again by low effectiveness. The effectiveness of spoiler 3 was not quite as nonlinear as the effectiveness of the combination of spoilers 2 and 3; therefore, the nonlinearity could probably be eliminated by carefully programmed projection rates and/or extent of spanwise segment used. Adding the deflector to the particular spoiler system used herein alleviated the initial low effectiveness; however, the effectiveness remained nonlinear with projection.

Spoiler or spoiler-deflector projection resulted in adverse yaw for all the moderate to high angle-of-attack ranges. The yawing moments produced by the control projection were not too large, however, to be controlled by a normal rudder installation.

It appears, from the data and analysis presented, that spoilers or spoiler-deflector combinations might possibly be used as a low-speed lateral-control device on a blowing boundary-layer-control configuration. In order to obtain the desired roll response, however, development work will be required for each configuration under consideration.

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Pressure-Distribution Characteristics

While the regular force tests of the model were being conducted, considerable surface-pressure-distribution data were also obtained. All of these pressure-distribution data are presented in tables 1 to 23, but only the typical and most pertinent data will be presented for discussion in the present paper.

Chordwise pressure distributions.- The chordwise pressure distributions at spanwise station 6 $\left(\frac{y}{b/2} = 0.800\right)$ are presented in figure 21 for the basic wing and for full-span trailing-edge-flap configurations with and without boundary-layer control. The data are presented for an angle of attack near maximum lift in each case. Boundary-layer control is seen to increase the loading over the whole chord with very high peak loading conditions near the leading- and trailing-edge flap hinge lines, as indicated by the magnitude of the pressure coefficient, C_p.

Chordwise loadings of the fuselage at spanwise stations 1 and 2 are shown in figure 22. The test conditions of the data of figure 22 correspond with those presented for the wing in figure 21. The wing is seen to have a very large influence on the fuselage pressures in the vicinity of the wing. Because the fuselage is circular in cross section, not uniform in diameter, and much longer in chord than the wing, the chordwise pressures could not be summed in the normal manner of integrating the pressure coefficients along the chord with these summations being directly comparable to the wing pressures. The fuselage pressures must be weighted because of the very long chord lengths and the variable spanwise locations of the orifices of a particular station (see fig. 5). This weighting of the fuselage pressures was necessary for determining the span-loading characteristics of the whole configuration. There are several ways in which the fuselage pressures could be weighted, but the one selected herein is described fully in appendix A.

The effect of aileron deflection on the chordwise loading at station 6 for half- and full-span flap configurations with boundary-layer control is shown in figure 23. Aileron deflection primarily affected only the aileron and the portion of the wing just forward of the aileron. With boundary-layer-control air blowing over the aileron very high peak negative pressures occurred over the aileron nose radius when the aileron was deflected downward.

The effect of deflection of spoiler 3 and spoiler deflector 3 on the chordwise loading at station 6 for the full-span flap configuration only is shown in figure 24. These lateral-control devices are seen to have a similar effect on the loading; that is, the loading over a considerable portion of the wing was greatly reduced both forward and aft of the control location.

<u>Span-loading characteristics</u>.- The span-loading characteristics of several half- and full-span flap configurations are shown in figures 25 and 26. The curves of figure 25 show span loadings of configurations with and without boundary-layer control while the curves of figure 26 show the change in span loadings resulting from aileron deflection on half- and full-span flap configurations with boundary-layer control. The loading points at $\frac{y}{b/2}$ of 0 and 0.154, as pointed out previously, were weighted according to the method described in appendix A.

Without blowing over the ailerons (fig. 25(a)) a rather abrupt change in loading is noted in the vicinity of the flap-aileron juncture $\left(\frac{y}{b/2} = 0.693\right)$. The loading over the outboard (aileron) portion of the wing (fig. 25(a)) is considered to be normal; however, blowing over the inboard (flap) portion of the wing greatly increased the loading over that portion (figs. 25(a) and (c)). Drooping the ailerons and applying blowing (fig. 25(b)) greatly increased the loading over the aileron portion of the wing and further increased the loading of the flapped portion. The large loading change at the flap-aileron juncture was also eliminated. Drooping the ailerons of the configuration without boundary-layer control (fig. 25(c)) produced a smaller but similar result to that obtained with aileron deflection and blowing.

The span-loading characteristics of half- and full-span flap configurations with boundary-layer control and aileron deflection are shown in figure 26. Aileron deflection is seen to have a large influence on the loading as might have been expected from results of the rolling-moment data previously discussed. Downward deflection of the aileron (fig. 26(c)) is seen to result in a high loading configuration, even for the half-span blowing flap configuration.

Performance Calculations

Landing performance with and without boundary-layer control.- The landing performance of the configurations with and without boundary-layer control was calculated by the methods described in detail for two configurations in appendix B. The basic trim data (fig. 27) on which the calculations were based were obtained from the longitudinal-control data of figure 12. It was assumed that a high-lift tail was used for trim.

The landing-flare calculations of the airplane without boundary-layer control utilized what might be considered a normal landing procedure of a jet airplane; that is, the landing configuration (flap setting, drag device, and power setting) was established during the approach and was not changed until the end of the runway was reached. The variables used during the flare were the angle of attack and the power condition. The only limiting condition of the angle-of-attack variations was that angle of attack would regulate speed from a value of $1.30V_{stall}$ at the initiation of the flare to a value of $1.15V_{stall}$ at touchdown. The power was shut off after the approach end of the runway was reached, and the flare was continued until the touchdown. At touchdown a drag device (assumed to be a drag parachute in the present case) having a wing drag coefficient of 0.12 was used during the ground roll.

The landing-flare procedure assumed for the airplane with boundary-layer control was somewhat unconventional. The angle of attack was varied in a conventional manner to obtain $1.30V_{stall}$ and $1.15V_{stall}$ for the approach and touchdown conditions, respectively, but a drag device producing an arbitrary amount of drag was used at the initiation of the flare while the flap setting and power condition used during the approach was maintained. Without the use of some additional drag during the flare, preliminary calculations showed that the airplane floating tendency resulting from the power setting required for the approach configuration with boundary-layer control would cause the airplane to have a very long stretchout of the flare. This stretchout of the flare could result in a distance to touchdown over a 50-foot obstacle much longer than that of a configuration without boundary-layer control. It should be noted that the effect caused by an increase in drag during the flare could have been accomplished by a reduction in engine thrust by an amount comparable to the assumed increase in drag, provided the engine could produce sufficient bleed-air for boundary-layer control at the reduce thrust condition.

The results of the landing performance calculations for a wing loading of 60 are shown graphically in figure 28. The configuration without boundary-layer control (fig. 28(a)) is seen to travel a total

distance during landing of about 3,900 feet, while the configuration with boundary-layer control having an arbitrary drag coefficient of 0.06 added at the beginning of the flare (fig. 28(b)) traveled about 4,100 feet. The floating tendency of the airplane with boundary-layer control is very noticeable in that the distance to touchdown over the 50-foot obstacle was about 13.5 percent greater than that of the configuration without boundary-layer control. The ground roll of the airplane with boundary-layer control was shorter than that of the airplane without boundary-layer control because of a lower touchdown speed and because, when the engine power was shutoff at touchdown, the airplane reverted to a low-lift configuration without boundary-layer control which would result in a large increase in weight on the wheels. This, of course, would provide better braking characteristics.

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In order to determine the effect on the landing characteristics of the airplane with boundary-layer control of adding more drag at the beginning of the flare, calculations were made for a drag coefficient increase of 0.12. The results of these calculations are shown in figure 28(c). The total landing distance of this configuration was only about 3,200 feet which was about 19 percent shorter than the configuration without boundary-layer control and about 23 percent less than the other configuration with boundary-layer control.

Take-off performance with and without boundary-layer control.- Because of the straightforward manner in which the take-off distances are normally calculated (ref. 5, for example) no detailed calculations in appendix form will be presented. The basic assumptions and general results of the calculations will, however, be discussed.

The take-off calculations were considered in two parts: (1) the ground roll to obtain the lift-off velocity $\left(1.15V_{stall}\right)$ and (2) the distance to clear a 50-foot obstacle after lift-off. The velocity corresponding to $1.15V_{stall}$ was that used in reference 5 and is not necessarily the optimum lift-off speed.

It is readily apparent in the formulas presented in reference 5 that the shortest distance to lift-off velocity will be accomplished by the configuration with the greatest thrust and the lowest drag. For the present tests this thrust-drag requirement was met by the basic unflapped configuration. The distance to obtain the desired lift-off velocity using the basic configuration at $\alpha = 0^{\circ}$ having a wing loading of 60 was approximately 1,700 feet. For comparison, if the flapped configuration without boundary-layer control $\left(\delta_{f,a} = 37^{\circ},\ \delta_n = 30^{\circ}\right)$ had been used for the ground roll instead of the basic unflapped configuration, the total distance to obtain lift-off velocity would have been increased about 13 percent.

When the velocity for lift-off is reached, the airplane is assumed to be quickly converted to the desired high-lift configuration while at the same time the aircraft is rotated to the best climb angle as determined by the external forces on the aircraft (thrust, drag, and weight). The distance from this point to clear a 50-foot obstacle was then assumed to be equal to the relationship, 50 ft/tan (climb angle). In the present case for the configuration without boundary-layer control this distance was about 290 feet. Neglecting the transition distance and time between ground roll and climb, the distance from $V = 0$ to clear a 50-foot obstacle for the configuration without boundary-layer control was about 2,000 feet.

For the configuration with boundary-layer control the distance to lift-off velocity was shortened somewhat because the lift-off speed for the high-lift configuration with boundary-layer control was lower than that of the configuration without boundary-layer control (183 ft/sec as compared with about 213 ft/sec). The distance from $V = 0$ to lift-off speed was about 1,100 feet for the configuration with boundary-layer control, and the distance to clear a 50-foot obstacle was about 370 feet. The total distance to clear a 50-foot obstacle was, therefore, about 1,500 feet which was about 25 percent less distance than that required for the configuration without boundary-layer control. The boundary-layer-control calculations included an assumed 8-percent thrust loss resulting from boundary-layer control air bleed.

CONCLUSIONS

Tests conducted in the Langley full-scale tunnel to determine the effects of blowing boundary-layer control on the aerodynamic characteristics of a large-scale, unswept fighter-type airplane model indicates the following results:

1. Wing leading-edge blowing in combination with large values of wing leading-edge-flap deflection was a very effective leading-edge flow-control device for wings having highly loaded trailing-edge flaps.

2. With leading-edge blowing applied, there was no hysteresis of the lift, drag, and pitching-moment characteristics upon recovery from stall.

3. End plates were found to improve the lift and drag characteristics of the test configuration in the moderate angle-of-attack range.

4. Blockage up to one-quarter of the blowing-slot area was not detrimental to the aerodynamic characteristics.

5. Blowing boundary-layer control resulted in a considerably reduced landing speed and reduced landing and take-off distances.

6. Ailerons were very effective lateral-control devices when used with blowing flaps.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., April 7, 1960.

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APPENDIX A

METHOD USED FOR COMPUTING SECTION NORMAL-FORCE COEFFICIENTS ON THE FUSELAGE

In order to calculate the normal force acting on the fuselage, static pressures over the fuselage were measured by surface orifices located at various longitudinal stations. The fuselage had a circular L
cross section at all longitudinal stations as shown in figure 5. The 9
orifices were placed every 60° around one side of the fuselage as shown 2
in the cross-sectional view A-A of figure 5. For purposes of computing 7
the forces on the fuselage by using electronic computers, the pressure at an orifice is assumed to act over an area which extends half-way to the next orifice as shown by the shaded area, Δx by Δy, projected on the horizontal plane in figure 5. The total normal force F_n on one station (longitudinal row of orifices) of the fuselage can, therefore, be written as:

$$F_n = \sum_{i=1}^{i=n} p_{i,l} \Delta x_i \Delta y_i - \sum_{i=1}^{i=n} p_{i,u} \Delta x_i \Delta y_i \qquad (1)$$

where the subscripts $1, 2, \ldots n$ refer to the pressure orifice number, and the subscripts l and u refer to the lower and upper surfaces of the fuselage, respectively.

Adding and subtracting the following expression to the right-hand side of equation (1):

$$\sum_{i=1}^{i=n} p_{\infty} \Delta x_i \Delta y_i$$

results in the following equation:

$$F_n = \sum_{i=1}^{i=n} \left(p_{i,l} - p_{\infty}\right)\Delta x_i \Delta y_i - \sum_{i=1}^{i=n} \left(p_{i,u} - p_{\infty}\right)\Delta x_i \Delta y_i \qquad (2)$$

Dividing equation (2) by the dynamic pressure q_∞ and by the total area of all the horizontal projections of the individual orifice areas ΔA where $\Delta A = \sum_{k=1}^{k=n} \Delta x_k \, \Delta y_k$ results in the following equation:

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$$\frac{F_n}{q_\infty \Delta A} = c_n = \sum_{i=1}^{i=n} \left(\frac{p_{i,l} - p_\infty}{q_\infty}\right)\frac{\Delta x_i \, \Delta y_i}{\Delta A} - \sum_{i=1}^{i=n} \left(\frac{p_{i,u} - p_\infty}{q_\infty}\right)\frac{\Delta x_i \, \Delta y_i}{\Delta A} \tag{3}$$

By substituting C_p for $\left(\frac{p - p_\infty}{q_\infty}\right)$ and letting $\frac{\Delta x_i \, \Delta y_i}{\Delta A} = IF_{c_{n,i}}$ (integrating factor), the equation for c_n becomes:

$$c_n = \sum_{i=1}^{i=n} IF_{c_{n,i}} C_{p,i,l} - \sum_{i=1}^{i=n} IF_{c_{n,i}} C_{p,u,i} \tag{4}$$

On the wing the Δy_k values are constant and are equal to Δy_i. The wing integrating factor then reduces to:

$$IF_{c_{n,i}} = \frac{\Delta x_i \, \Delta y_i}{\sum_{k=1}^{k=n} \Delta x_k \, \Delta y_k} = \frac{\Delta x_i}{\sum_{k=1}^{k=n} \Delta x_k} = \frac{\Delta x_i}{c}$$

Since the fuselage used in these tests has a circular cross section and orifices placed at a constant angular distance around the fuselage, it can be seen from the cross-sectional view of figure 5 that the horizontal projection Δy for the inboard row of orifices is equal to $r\left(\frac{\sin 60^\circ}{2}\right)$, and for the outboard row Δy is equal to $r\left(1 - \frac{\sin 60^\circ}{2}\right)$, where r is the radius of the fuselage at the particular orifice location. The integrating constant $IF_{c_{n,i}}$, however, is the same for either the inboard or outboard row of orifices, that is:

$$\left(IF_{c_{n,i}}\right)_{inboard} = \frac{\Delta x_i \ \Delta y_i}{\Delta A}$$

$$= \frac{\Delta x_i \ \Delta y_i}{\sum_{k=1}^{k=n} \Delta x_k \ \Delta y_k}$$

$$= \frac{\Delta x_i r_i \ \frac{\sin 60^\circ}{2}}{\sum_{k=1}^{k=n} \Delta x_k r_k \ \frac{\sin 60^\circ}{2}}$$

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$$= \frac{\Delta x_i r_i}{\sum_{k=1}^{k=n} \Delta x_k r_k}$$

$$= \frac{\Delta x_i r_i}{\text{Horizontal projection of the fuselage area}}$$

and

$$\left(IF_{c_{n,i}}\right)_{outboard} = \frac{\Delta x_i r_i \left(1 - \frac{\sin 60^\circ}{2}\right)}{\sum_{k=1}^{k=n} \Delta x_k r_k \left(1 - \frac{\sin 60^\circ}{2}\right)}$$

$$= \frac{\Delta x_i r_i}{\sum_{k=1}^{k=n} \Delta x_k r_k}$$

$$= \frac{\Delta x_i r_i}{\text{Horizontal projection of the fuselage area}}$$

For the span-loading plots of $c_n \frac{c}{c_{av}}$ against $\frac{y}{b/2}$ the $\frac{c}{c_{av}}$ term for the fuselage stations must be calculated so that the term $\left(c_n \frac{c}{c_{av}}\right)_{fuselage}$ can be summed directly with the wing loading term, and thus the overall configuration load can be determined.

In the usual manner, the ordinate $c_n \frac{c}{c_{av}}$ represents $\frac{dC_N}{d\left(\frac{y}{b/2}\right)}$ and the abscissa is $\frac{y}{b/2}$. The total normal-force coefficient C_N is then equal to the area under the curve derived from the above ordinate and abscissa:

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$$C_N = \int_0^{1.0} \frac{dC_N}{d\left(\frac{y}{b/2}\right)} d\left(\frac{y}{b/2}\right) \tag{5}$$

Since only one-half of the model wing area is being considered in the pressure-distribution work (i.e., $\frac{y}{b/2}$ from the fuselage center line to the wing tip):

$$dC_N = \frac{dF_N}{q_\infty S/2}$$

where F_N is the normal force and $S/2$ is one-half of the total wing area. Then as an approximation, finite increments of span are used at each spanwise orifice station:

$$\Delta F_N = c_n q_\infty \Delta A$$

and

$$\frac{dC_N}{d\left(\frac{y}{b/2}\right)} \approx \frac{\Delta F_N}{q_\infty (S/2)\frac{\Delta y}{b/2}}$$

$$= \frac{c_n q_\infty \Delta A}{q_\infty \left(\frac{b}{2} c_{av}\right)\frac{\Delta y}{b/2}}$$

$$= \frac{c_n \sum_{k=1}^{k=n} \Delta x_k \Delta y_k}{c_{av} \Delta y} \qquad (6)$$

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Normally (i.e., for the wing stations) Δy_k is constant and is equal to Δy which results in the following:

$$\frac{dC_N}{d\left(\frac{y}{b/2}\right)} \approx \frac{c_n \sum_{k=1}^{k=n} \Delta x_k \Delta y_k}{c_{av} \Delta y}$$

$$= \frac{c_n \sum_{k=1}^{k=n} \Delta x_k}{c_{av}}$$

$$= c_n \frac{c}{c_{av}} \qquad (7)$$

This formula, however, should not be used for the fuselage. If the total fuselage length were used as c in formula (7), the fuselage pressure data would be weighted too heavily as compared with the wing data because the c/c_{av} term of equation (7) assumes a constant, finite spanwise dimension; whereas, the fuselage stations do not have a constant spanwise dimension. The fuselage chord was therefore foreshortened by an amount which was proportional to the actual pressure area involved; that is, an equivalent chord length was used for the fuselage.

$$\left({}^{c}/c_{av}\right)_{fuselage} = \frac{\sum_{k=1}^{k=n} \Delta x_k \, \Delta y_k}{c_{av} \, \Delta y}$$

In the case of the circular cross-section fuselage used in these tests with orifices placed at 60° intervals around the side of the fuselage:

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$$\left({}^{c}/c_{av}\right)_{fuselage} = \frac{\sum_{k=1}^{k=n} \Delta x_k r_k \frac{\sin 60^{\circ}}{2}}{c_{av} r_{max} \frac{\sin 60^{\circ}}{2}}$$

(where r_{max} is the maximum fuselage radius and the lateral distance over which the fuselage loading is assumed to extend)

$$\left({}^{c}/c_{av}\right)_{fuselage} = \frac{\sum_{k=1}^{k=n} \Delta x_k r_k}{c_{av} r_{max}}$$

$$= \frac{\text{Horizontal projection of the fuselage area}}{c_{av} r_{max}} \qquad (8)$$

Formula (8) applies to both of the fuselage stations.

APPENDIX B

LANDING PERFORMANCE WITH AND WITHOUT BOUNDARY-LAYER CONTROL

The landing performance calculations were made, for comparison, for two configurations: (1) $\delta_n = 30^\circ$, $\delta_{f,a} = 37^\circ$ without boundary-layer control, and (2) $\delta_n = 50^\circ$, $\delta_{f,a} = 47^\circ$, $C_{\mu,k} = 0.010$, $C_{\mu,f} = 0.012$, $C_{\mu,a} = 0.004$. The approach and landing velocities were considered to be $1.30V_{stall}$ and $1.15V_{stall}$, respectively, for each configuration. The force data used for the calculations were assumed to be for a trimmed condition having a wing loading W/S of 60. The trim data shown in figure 27 were derived from the tail effectiveness data of figure 12. An increment of drag coefficient of 0.06 was arbitrarily added to all the drag data to account for the drag of the landing gear and other protuberances. For the boundary-layer-control configuration two calculations were made. The first had an increment of drag coefficient of 0.06 added at the initiation of the flare, and the other used an increment of 0.12. An addition of drag was required at the initiation of the flare to reduce speed so that angle of attack could be increased to a value, at touchdown, corresponding to approximately $1.15V_{stall}$. In order to use the lift capability of the boundary-layer-control configuration without the increase in drag, the horizontal distance covered during the flare would have been prohibitive. The reason for the flare problem of the boundary-layer-control configuration is the assumption of an essentially constant power setting for boundary-layer control. This power setting keeps the airplane essentially in equilibrium; therefore, an increase in angle of attack would arrest the rate of sink and would result in a stretchout of the flare maneuver.

For the thrust required to maintain equilibrium during the steady-state approach condition, a calculation was made to determine the approximate thrust loss resulting from the use of sufficient bleed air for boundary-layer control. The performance calculations were for a turbojet engine. From these calculations it was determined that approximately an 8-percent thrust loss would be incurred in the landing approach because of the boundary-layer-control bleed. This would be no particular problem for the configuration under consideration, however, because sufficient excess thrust would still be available for an aborted landing. The thrust loss resulting from boundary-layer-control bleed was not included in the landing-flare calculations.

Further assumptions made for the landing-performance calculations were: (1) the approach angle was 3°, (2) the throttle setting was held constant, at the previously determined approach setting, until such time that the engine was shut off after the end of the runway was reached, and (3) speed was reduced in the flare by increasing angle of attack at a rate required for a smooth flight path with approximately no excess forward speed, above $1.15V_{stall}$ nor excess sinking speed, above 3.0 ft/sec, at touchdown.

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The conditions for the steady-state approach speed (initial conditions for the flare calculations) were determined as follows:

$$V_A = \text{Velocity of approach} = 1.30V_{stall}$$

$$C_{L,A} = \text{Approach lift coefficient} = C_{L,max}/1.30^2$$

For this value of $C_{L,A}$ a comparable value of $C_{D,A}$ exists. A flight-path angle γ was selected (3° in present case) from the examination of a flight-path equilibrium diagram:

$$\gamma = \tan^{-1}\left(\frac{D}{L}\right)$$

or

$$\frac{L}{D} = \frac{1}{\tan \gamma}$$

which is the value for the equilibrium condition. Also

$$\frac{L}{D} = \frac{C_L + T_c' \sin \alpha}{C_D - T_c' \cos \alpha}$$

where

$$T_c' = \frac{\text{Thrust}}{qS}$$

Therefore,

$$T_c' = \frac{C_L - \frac{L}{D} C_D}{-\frac{L}{D} \cos \alpha - \sin \alpha}$$

The resultant lift coefficient $C_{L,R}$ along the glide path is equal to $C_{L,A} + T_c' \sin \alpha$, and the resultant dynamic pressure q_R is equal to $\frac{W/S}{C_{L,R}}$ (where W = Airplane weight).

$$T_{req} = T_c' q_R S$$

These calculations will furnish the initial data for determining the landing-flare characteristics.

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The landing-flare formulas and calculations for the configurations with and without boundary-layer control are given in detail in tables 24 to 27. The small increase in thrust as speed decreased (tables 26 and 27) is a characteristic of the engine. The reason for the horizontal and vertical acceleration not being zero for the initial condition of each configuration (tables 26 and 27) is attributed to the small inaccuracy of the thrust value. Actually, the small number of decimal places to which the data were computed would preclude the acceleration values being zero. A plot of the landing flare of the two configurations is given as figure 27.

The ground-roll distance was determined by using the method outlined as follows:

$$\text{Ground roll} = \frac{W}{2g} \cdot \frac{V_L^2}{D - T + G}$$

where

W	airplane weight, lb
V_L	landing velocity, ft/sec
D	drag, at $0.7V_L$ and for angle of attack at touchdown, lb
T	engine thrust, lb
g	acceleration due to gravity
G	mean ground braking force, lb

$$G = k(W - L)$$

where

k friction coefficient, assumed to be 0.25

L lift at $0.7V_L$ and for angle of attack at touchdown

In the present case, thrust was assumed to be zero for the ground roll.

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TABLE 1

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = -.5°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.270	.304	.271	.290
.053	.057	.067	.064	.055
.100	-.070	-.051	-.093	-.097
.145	-.053	-.093	-.080	-.059
.189	.016	-.055	-.013	-.008
.234	-.029	.021	.025	-.013
.280	-.025	.021	-.013	-.013
.326	.016	-.004	-.013	.000
.371	-.016	.025	-.030	.013
.392	.025	-.004	.017	.050
.413	-.016	.042	.000	.055
.434	-.008	.089	-.042	.004
.457	.004	.000	-.034	-.029
.480	-.008	-.004	-.055	-.092
.502	-.037	.000	-.055	-.118
.551	-.016	.000	-.059	-.097
.585	-.008	-.025	-.047	-.080
.592	-.008	.008	-.042	-.042
.613	.012	.025	-.025	-.038
.634	.020	.025	.000	-.013
.655	.029	.034	.021	.013
.675	.045	.008	.051	.046
.696	.065	.021	.047	.046
.774	.074	.093	.034	.034
.852	-.037	-.008	-.038	-.088
.930	.041	.030	.030	.017

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.058	-.017	.000	.131	.131
	.080	-.004	-.013	-.013	.000	.025
	.130	-.041	-.059	-.055	-.034	-.057
	.145	-.191	-.076	-.097	-.059	-.033
	.155	-.058	-.075	-.021	-.017	.008
	.180	-.145	-.063	-.050	-.021	-.025
	.220	-.083	-.072	-.071	-.059	-.037
	.270	-.066	-.084	-.134	-.059	-.061
	.400	-.099	-.122	-.139	-.131	-.090
	.620	-.083	-.093	-.084	-.068	-.049
	.685					
	.693					
	.700	.017	-.013	-.042	.004	-.041
	.720	-.029	-.034	-.055	-.025	-.012
	.750	-.029	-.038	-.055	-.051	-.029
	.800	-.025	-.025	-.034	-.038	-.016
	.900	.000	-.008	-.008	-.017	.000
	.980	.075	.084	.071	.072	.086
Lower	.025	.182	.156	.092	.030	.029
	.120	.075	.055	-.004	.042	.176
	.220	-.021	-.042	-.118	-.085	.000
	.300	-.087	-.118	-.143	-.102	-.061
	.620	-.058	-.072	-.118	-.080	-.057
	.750	-.062	-.067	-.105	-.106	-.070
	.850	-.025	-.025	-.055	-.059	-.016
	.950	.054	.055	.038	.034	.033

α = 10.7°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.004	.546	.042	.356
.053	-.167	.328	-.148	.110
.100	-.179	.151	-.250	-.114
.145	-.128	.063	-.195	-.085
.189	-.060	.101	-.136	-.051
.234	-.090	.151	-.047	-.072
.280	-.064	.151	.004	-.055
.326	-.064	.147	.008	-.034
.371	-.150	.189	-.093	.042
.392	.004	.025	-.313	.233
.413	-.235	.227	-.686	.330
.434	-.278	.223	-.873	.297
.457	-.286	.021	-.868	.237
.480	-.282	.021	-.665	.152
.502	-.308	.021	-.500	.102
.551	-.209	.021	-.258	.034
.585	-.167	.071	-.182	.034
.592	-.137	.118	-.140	.025
.613	-.073	.118	-.089	.008
.634	-.056	.109	-.047	.025
.655	-.047	.101	-.013	.038
.675	-.004	.050	.017	.068
.696	.030	.084	.059	.068
.774	.060	.134	.055	.072
.852	-.047	.017	-.025	.050
.930	.038	.042	.047	.038

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-1.221	-.899	-.771	-.754	-.803
	.080	-1.255	-.882	-.822	-.801	-.829
	.130	-1.174	-.907	-.843	-.843	-.859
	.145	-1.127	-.882	-.822	-.818	-.850
	.155	-1.127	-.915	-.839	-.822	-.850
	.180	-1.089	-.890	-.851	-.839	-.863
	.220	-1.021	-.894	-.851	-.839	-.868
	.270	-.876	-.890	-.839	-.851	-.859
	.400	-.570	-.731	-.746	-.767	-.739
	.620	-.204	-.378	-.449	-.424	-.466
	.685					
	.693					
	.700	-.111	-.252	-.313	-.360	-.397
	.720	-.098	-.244	-.313	-.364	-.380
	.750	-.098	-.210	-.284	-.326	-.363
	.800	-.064	-.168	-.220	-.267	-.329
	.900	-.021	-.084	-.148	-.186	-.261
	.980	.043	-.034	-.093	-.127	-.167
Lower	.025	.787	.731	.699	.674	.568
	.120	.485	.453	.398	.419	.397
	.220	.289	.265	.220	.225	.179
	.300	.183	.172	.152	.144	.094
	.620	.047	.034	.000	-.004	-.111
	.750	-.013	-.017	-.068	-.089	-.111
	.850	-.004	-.013	-.059	-.072	-.098
	.950	.034	-.013	-.042	-.064	-.124

α = 14.6°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.065	.649	-.082	.277
.053	-.216	.426	-.247	.074
.100	-.155	.219	-.338	-.156
.145	-.095	.131	-.273	-.156
.189	-.026	.160	-.212	-.082
.234	-.039	.211	-.082	-.113
.280	-.013	.228	-.013	-.091
.326	.000	.207	-.013	-.052
.371	-.078	.245	-.078	.056
.392	.009	.255	-.169	.282
.413	-.185	.278	-.528	.377
.434	-.267	.270	-.524	.351
.457	-.328	.240	-.550	.295
.480	-.405	.210	-.567	.191
.502	-.474	.180	-.593	.126
.551	-.448	.150	-.645	.026
.585	-.448	.130	-.676	.000
.592	-.448	.127	-.650	-.489
.613	-.285	.097	-.580	-.039
.634	-.323	.097	-.481	-.056
.655	-.237	.084	-.351	-.074
.675	-.135	.034	-.243	-.065
.696	-.065	.059	-.121	-.065
.774	.086	.152	.030	.022
.852	.013	.004	.000	-.100
.930	.034	.059	.013	.017

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.582	-.523	-.511	-.602	-.638
	.080	-.599	-.561	-.541	-.637	-.625
	.130	-.590	-.574	-.563	-.628	-.638
	.145	-.595	-.578	-.567	-.637	-.638
	.155	-.590	-.565	-.559	-.637	-.625
	.180	-.590	-.607	-.572	-.632	-.638
	.220	-.595	-.595	-.589	-.654	-.634
	.270	-.586	-.628	-.615	-.663	-.655
	.400	-.645	-.666	-.671	-.723	-.677
	.620	-.683	-.679	-.745	-.671	-.690
	.685					
	.693					
	.700	-.595	-.603	-.645	-.723	-.690
	.720	-.607	-.603	-.680	-.723	-.681
	.750	-.578	-.590	-.693	-.723	-.690
	.800	-.489	-.548	-.654	-.689	-.673
	.900	-.358	-.409	-.567	-.628	-.638
	.980	-.219	-.312	-.446	-.524	-.569
Lower	.025	.831	.776	.749	.706	.604
	.120	.557	.510	.446	.468	.431
	.220	.350	.329	.247	.256	.198
	.300	.245	.215	.178	.160	.116
	.620	.038	.008	-.048	-.052	-.147
	.750	-.055	-.072	-.139	-.160	-.194
	.850	-.080	-.097	-.165	-.199	-.220
	.950	-.114	-.186	-.251	-.303	-.345

TABLE 1 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 18.6 °

x/l	Fuselage: 0.000, Upper surface	Fuselage: 0.000, Lower surface	Fuselage: 0.154, Upper surface	Fuselage: 0.154, Lower surface
.032	-.121	.738	-.194	.239
.053	-.269	.541	-.345	.053
.100	-.156	.300	-.392	-.195
.145	-.113	.215	-.336	-.191
.189	-.076	.240	-.267	-.129
.234	.[illegible]	.279	-.121	-.155
.280	.013	.296	-.009	-.115
.326	.043	.270	-.004	-.075
.371	-.048	.322	-.013	.053
.392	-.004	.328	-.142	.337
.413	-.225	.335	-.431	.452
.434	-.321	.331	-.423	.425
.457	-.364	.330	-.431	.368
.480	-.407	.250	-.444	.266
.502	-.485	.180	-.461	.177
.551	-.520	.130	-.543	.044
.585	-.554	.130	-.578	.009
.592	-.593	.129	-.604	-.589
.613	-.450	.103	-.634	-.071
.634	-.511	.077	-.608	-.115
.655	-.433	.052	-.569	-.168
.675	-.312	-.004	-.492	-.191
.696	-.230	.026	-.371	-.230
.774	-.009	.124	-.091	-.137
.852	.026	-.009	-.017	-.195
.930	.035	.082	.004	.[illegible]

Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.440	-.434	-.408	-.466	-.559
	.080	-.453	-.455	-.434	-.492	-.576
	.130	-.457	-.455	-.452	-.492	-.580
	.145	-.457	-.464	-.461	-.492	-.576
	.155	-.444	-.451	-.452	-.487	-.567
	.180	-.449	-.459	-.443	-.483	-.580
	.220	-.453	-.459	-.452	-.492	-.580
	.270	-.466	-.472	-.479	-.513	-.598
	.400	-.500	-.532	-.532	-.573	-.650
	.620	-.577	-.622	-.620	-.638	-.702
	.685					
	.693					
	.700	-.568	-.597	-.580	-.686	-.719
	.720	-.598	-.614	-.620	-.690	-.710
	.750	-.611	-.610	-.647	-.686	-.706
	.800	-.594	-.601	-.638	-.681	-.706
	.900	-.517	-.549	-.625	-.677	-.706
	.980	-.444	- 434	-.572	-.647	-.680
Lower	.025	.897	.837	.793	.750	.624
	.120	.624	584	.514	.509	.459
	.220	.427	391	.332	.328	.225
	.300	.308	266	.257	.216	.147
	.620	.047	013	-.035	-.043	-.165
	.750	-.068	-.107	-.168	-.185	-.230
	.850	-.141	-.167	-.239	-.254	-.282
	.950	-.226	-.275	-.354	-.397	-.433

TABLE 2

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 30°; δ_f = 37°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
α = -1.0°											
.032	.265	.306	.291	.297	Upper	.010	.774	.729	.703	.739	.731
.053	.051	.069	.064	.059		.080	.231	.129	.114	.115	.158
.100	-.085	-.039	-.098	-.085		.130	-.329	-.539	-.572	-.534	-.543
.145	-.090	-.116	-.073	-.051		.145	-2.141	-2.259	-1.944	-2.081	-1.902
.189	-.004	-.052	-.013	.000		.155	-.944	-1.074	-1.059	-.944	-.791
.234	-.038	.047	.004	.004		.180	-.645	-.694	-.712	-.658	-.620
.280	-.030	.043	-.004	.008		.220	-.423	-.522	-.534	-.534	-.436
.326	-.004	.034	.009	.025		.270	-.329	-.453	-.517	-.397	-.363
.371	-.038	.047	.150	.068		.400	-.338	-.418	-.436	-.397	-.295
.392	.013	.004	.047	.034		.620	-.423	-.509	-.559	-.013	-.197
.413	-.068	.091	-.056	-.110		.685					
.434	-.090	.138	-.192	.000		.693					
.457	-.098	.000	-.244	.110		.700	-.346	-.457	-1.050	-.226	-.150
.480	-.141	.004	-.244	.157		.720	-.355	-.461	-.974	-.274	-.150
.502	-.188	.004	-.261	.165		.750	-.350	-.470	-1.059	-.278	-.150
.551	-.197	.004	-.286	.220		.800	-.355	-.487	-1.004	-.261	-.145
.585	-.201	.194	-.312	.241		.900	-.432	-.543	-.771	-.201	-.107
.592	-.201	.216	-.316	-.110		.980	-.440	-.517	-.534	-.107	-.026
.613	-.167	.164	-.286	.072	Lower	.025	-.944	-.164	-.203	-.188	-.201
.634	-.162	.091	-.274	-.313		.120	-.137	-.302	-.254	-.248	-.201
.655	-.150	.000	-.265	-.483		.220	.329	-.185	-.271	-.252	-.192
.675	-.107	-.112	-.226	-.381		.300	.244	.034	-.131	-.252	-.188
.696	-.073	-.082	-.150	-.152		.620	.457	.457	.352	.051	-.094
.774	-.013	.091	-.013	.042		.750	.611	.582	.263	-.107	-.073
.852	-.085	-.013	-.056	-.059		.850	.363	.379	.313	-.171	-.073
.930	.004	.013	.013	.017		.950	.124	.082	.097	-.154	-.030
α = 10.2°											
.032	.009	.576	.035	.300	Upper	.010	-1.714	-1.374	-1.363	-1.359	-.674
.053	-.183	.354	-.150	.084		.080	-.840	-.736	-.794	-.891	-.692
.100	-.187	.160	-.282	-.146		.130	-1.497	-1.586	-1.632	-1.685	-1.627
.145	-.135	.053	-.234	-.119		.145	-4.103	-3.793	-3.286	-3.727	-3.472
.189	-.070	.102	-.168	-.079		.155	-2.097	-2.127	-2.077	-2.197	-1.862
.234	-.087	.151	-.053	-.084		.180	-1.418	-1.480	-1.486	-1.508	-1.327
.280	-.087	.160	.044	-.093		.220	-.962	-1.112	-1.107	-1.120	-.944
.326	-.096	.155	.044	-.110		.270	-.748	-.900	-.962	-.891	-.766
.371	-.165	.226	-.097	-.084		.400	-.587	-.691	-.723	-.715	-.613
.392	-.200	.266	-.357	.053		.620	-.587	-.598	-.670	-.384	-.483
.413	-.244	.301	-.578	.296		.685					
.434	-.274	.337	-.763	.419		.693					
.457	-.274	.340	-.644	.437		.700	-.544	-.403	-.878	-.406	-.435
.480	-.292	.335	-.520	.393		.720	-.535	-.390	-.812	-.437	-.431
.502	-.331	.325	-.481	.348		.750	-.500	-.403	-.869	-.445	-.418
.551	-.287	.315	-.450	.384		.800	-.505	-.417	-.675	-.410	-.400
.585	-.261	.306	-.468	.437		.900	-.496	-.479	-.618	-.326	-.344
.592	-.252	.310	-.516	-.309		.980	-.435	-.443	-.640	-.229	-.183
.613	-.174	.248	-.468	-.350	Lower	.025	.492	.585	.556	.556	.400
.634	-.183	.160	-.406	-.401		.120	.657	.687	.591	.604	.540
.655	-.170	.058	-.335	-.432		.220	.587	.585	.516	.516	.409
.675	-.109	-.062	-.238	-.269		.300	.479	.483	.428	.397	.270
.696	-.065	-.027	-.146	-.088		.620	.596	.625	.472	.084	-.078
.774	.065	.000	.000	.075		.750	.683	.740	.609	-.071	-.091
.852	-.078	.044	-.049	-.049		.850	.418	.470	.287	-.203	-.104
.930	.076	.035	.022	.035		.950	.178	.120	.022	-.229	-.117
α = 17.8°											
.032	-.122	.722	-.187	.195	Upper	.010	-4.279	-2.238	-2.153	-1.980	-2.050
.053	-.262	.536	-.370	.022		.080	-1.263	-2.331	-2.286	-2.119	-2.072
.100	-.170	.297	-.426	-.235		.130	-1.805	-2.122	-1.976	-1.788	-1.626
.145	-.140	.204	-.379	-.248		.145	-4.502	-1.821	-1.626	-1.510	-1.639
.189	-.057	.226	-.322	-.171		.155	-2.382	-1.706	-1.502	-1.410	-1.482
.234	-.061	.257	-.135	-.230		.180	-1.691	-1.604	-1.409	-1.336	-1.390
.280	-.079	.275	.074	-.239		.220	-1.228	-1.493	-1.329	-1.279	-1.281
.326	-.101	.279	.091	-.301		.270	-1.010	-1.400	-1.263	-1.223	-1.202
.371	-.236	.372	-.100	-.292		.400	-.765	-1.046	-1.063	-1.079	-1.045
.392	-.290	.400	-.648	-.066		.620	-.634	-.625	-.713	-.744	-.892
.413	-.345	.448	-.831	.448		.685					
.434	-.385	.443	-1.236	.558		.693					
.457	-.350	.440	-.988	.576		.700	-1.080	-.527	-.549	-.679	-.848
.480	-.350	.440	-.788	.523		.720	-.677	-.505	-.510	-.683	-.813
.502	-.372	.420	-.700	.479		.750	-.564	-.501	-.523	-.653	-.800
.551	-.306	.400	-.648	.456		.800	-.507	-.510	-.518	-.622	-.743
.585	-.271	.372	-.735	.523		.900	-.468	-.510	-.554	-.535	-.647
.592	-.262	.372	-.835	-.722		.980	-.415	-.448	-.496	-.466	-.507
.613	-.179	.315	-.683	-.580	Lower	.025	.778	.829	.780	.731	.590
.634	-.192	.235	-.522	-.390		.120	.765	.767	.700	.657	.520
.655	-.175	.129	-.374	-.434		.220	.691	.687	.634	.596	.450
.675	-.114	.004	-.270	-.288		.300	.573	.607	.567	.474	.337
.696	-.070	-.004	-.174	-.155		.620	.612	.687	.518	.170	-.087
.774	-.035	.124	-.013	.066		.750	.699	.771	.576	-.030	-.087
.852	-.087	.058	-.022	-.027		.850	.450	.514	.363	-.152	-.149
.930	.017	.080	.048	.053		.950	.249	.186	.137	-.296	-.262

TABLE 7 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 30°; δ_f = 37°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = 21.8°

x/l	Fuselage: 0.000, Upper surface	Fuselage: 0.000, Lower surface	Fuselage: 0.154, Upper surface	Fuselage: 0.154, Lower surface
.032	-.192	.298	-.358	.102
.053	-.300	.607	-.483	-.069
.100	-.174	.381	-.514	-.326
.145	-.139	.279	-.470	-.342
.189	-.031	.319	-.389	-.268
.234	-.022	.346	-.201	-.319
.280	-.040	.359	.054	-.319
.326	-.089	.368	.054	-.347
.371	-.268	.434	-.165	-.314
.392	.375	.470	-.672	.074
.413	-.452	.510	-.908	.499
.434	-.541	.510	-1.467	.610
.457	-.541	.500	-1.261	.643
.480	-.514	.480	-1.194	.587
.502	-.541	.450	-1.082	.536
.551	-.461	.420	-.908	.467
.585	-.447	.399	-.890	.300
.592	-.438	.394	-.970	-.703
.613	-.353	.310	-.845	-.580
.634	-.367	.213	-.657	-.435
.655	-.353	.106	-.523	-.448
.675	-.257	-.049	-.411	-.314
.696	-.201	-.066	-.313	-.227
.774	-.149	.097	-.085	-.018
.852	-.054	.040	-.036	-.055
.930	.013	.089	.022	.037

Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
Upper	.010	-1.786	1.529	-1.544	-1.328	-1.453
	.080	-1.865	1.684	-1.618	-1.445	-1.498
	.130	-1.662	1.639	-1.618	-1.391	-1.436
	.145	-1.586	1.617	-1.609	-1.382	-1.431
	.155	-1.493	1.489	-1.498	-1.292	-1.355
	.180	-1.458	1.449	-1.475	-1.257	-1.355
	.220	-1.387	1.414	-1.401	-1.279	-1.346
	.270	-1.347	1.400	-1.415	-1.292	-1.346
	.400	-1.094	1.254	-1.299	-1.284	-1.310
	.620	-.700	-.939	-1.054	-.908	-1.172
	.685					
	.693					
	.700	-1.139	-.868	-.897	-.979	-1.047
	.720	-.775	-.749	-.795	-.962	-1.015
	.750	-.638	-.696	-.735	-.926	-.997
	.800	-.563	-.634	-.689	-.877	-.966
	.900	-.501	-.558	-.610	-.756	-.894
	.980	-.394	-.465	-.536	-.657	-.783
Lower	.025	.784	.851	.814	.765	.613
	.120	.802	.802	.717	.662	.546
	.220	.736	.731	.684	.617	.470
	.300	.651	.643	.610	.505	.353
	.620	.647	.700	.550	.170	-.089
	.750	.731	.802	.430	-.031	-.121
	.850	.479	.545	.374	-.179	-.206
	.950	.270	.204	.139	-.344	-.403

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TABLE

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 30°$; $\delta_f = 37°$; $\delta_{a,L} = 37°$; $\delta_{a,R} = 37°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.000$ $C_{\mu,f} = 0.000$ $C_{\mu,a} = 0.000$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = -1.2°$											
.032	.267	.321	.267	.294	Upper	.010	.786	.705	.685	.707	.699
.053	.043	.081	.043	.047		.080	.201	.115	.081	.073	.112
.100	-.095	-.047	-.099	-.094		.130	-.359	-.538	-.617	-.586	-.621
.145	-.073	-.073	-.073	-.038		.145	-2.222	-2.252	-2.000	-2.177	-2.070
.189	-.013	-.034	-.022	-.004		.155	-.979	-1.073	-1.102	-1.013	-.888
.234	-.039	.047	.026			.180	-.671	-.688	-.745	-.703	-.690
.280	-.043	.060	-.026	.021		.220	-.436	-.517	-.566	-.569	-.492
.326	-.017	.047	-.013	.038		.270	-.350	-.449	-.540	-.453	-.418
.371	-.065	.077	-.026	.064		.400	-.342	-.415	-.477	-.466	-.414
.392	.009	-.121	.034	.055		.620	-.444	-.491	-.498	-.552	-.569
.413	-.069	.111	-.034	-.094		.685	-1.103	-.949	-1.055	-1.056	-1.673
.434	-.103	.154	-.198	.009		.693	-.803	-.744	-.762	-.819	-1.276
.457	-.112	-.021	-.233	.106		.700	-.376	-.423	-.485	-.573	-.824
.480	-.147	-.021	-.237	.153		.720	-.363	-.406	-.464	-.569	-.793
.502	-.207	-.021	-.259	.166		.750	-.363	-.419	-.485	-.569	-.798
.551	-.207	.021	-.298	.225		.800	-.372	-.423	-.481	-.586	-.802
.585	[illegible]	.201	-.328	.255		.900	-.444	-.457	-.506	-.617	-.811
.592	[illegible]	.231	-.332	-.302		.980	-.462	-.440	-.494	-.582	-.664
.613	-.164	.184	-.293	.055	Lower	.025	-.803	-.094	-.089	-.026	-.043
.634	-.168	.107	-.289	-.357		.120	-.094	-.188	-.119	-.060	-.065
.655	-.155	.009	-.267	-.489		.220	.359	-.107	-.140	-.091	-.078
.675	-.112	-.124	-.220	-.383		.300	.252	.073	-.017	-.082	-.095
.696	-.078	-.107	-.164	-.174		.620	.449	.470	.396	.319	.151
.774	-.052	.081	-.017	.047		.750	.607	.577	.459	.440	.354
.852	-.073	-.030	-.052	-.064		.850	.359	.376	.357	.405	.371
.930	.009	.021	.013	.017		.950	.111	.073	.068	.106	.125
$\alpha = 9.9°$											
.032	[illegible]	.565	.022	.309	Upper	.010	-1.901	-1.491	-1.632	-1.600	-1.473
.053	-.168	.362	-.155	.072		.080	-.861	-.723	-.859	-.862	-.794
.100	-.168	.137	-.279	-.130		.130	-1.521	-1.614	-1.753	-1.733	-1.773
.145	-.115	.062	-.235	-.134		.145	-4.135	-3.802	-3.430	-3.753	-3.595
.189	-.053	.101	-.177	-.085		.155	-2.115	-2.148	-2.223	-2.282	-2.016
.234	-.079	.150	-.066	-.103		.180	-1.416	-1.504	-1.619	-1.600	-1.473
.280	-.079	.159	.053	-.103		.220	-.979	-1.116	-1.225	-1.223	-1.081
.326	-.084	.159	.053	-.125		.270	-.747	-.922	-1.064	-.997	-.882
.371	-.161	.243	-.066	-.103		.400	-.603	-.719	-.836	-.846	-.776
.392	.071	.280	-.394	.040		.620	-.586	-.609	-.760	-.820	-.856
.413	-.238	.322	-.603	.313		.685	-1.329	-.706	-1.440	-1.458	-2.320
.434	-.273	.348	-.793	.425		.693	-1.101	-.512	-1.310	-1.254	-1.919
.457	-.273	.350	-.674	.452		.700	-.546	-.393	-.792	-.868	-1.160
.480	-.296	.350	-.545	.407		.720	-.516	-.388	-.729	-.877	-1.094
.502	-.344	.345	-.501	.362		.750	-.498	-.406	-.792	-.891	-1.116
.551	-.291	.340	-.456	.360		.800	-.494	-.428	-.662	-.895	-1.103
.585	-.256	.320	-.479	.434		.900	-.507	-.485	-.662	-.864	-1.098
.592	-.256	.326	-.536	-.425		.980	-.455	-.437	-.675	-.775	-1.006
.613	-.168	.260	-.492	-.420	Lower	.025	.520	.618	.622	.607	.529
.634	-.190	.281	-.443	-.429		.120	.686	.701	.644	.638	.529
.655	-.172	.088	-.154	-.461		.220	.594	.600	.577	.576	.485
.675	-.106	-.031	-.253	-.304		.300	.481	.517	.505	.487	.401
.696	-.066	-.026	-.164	-.140		.620	.590	.662	.640	.572	.269
.774	.018	.141	[illegible]	.058		.750	.691	.745	.684	.629	.556
.852	-.071	.162	-.058	.063		.850	.411	.468	.429	.390	.362
.930	.026	.044	.013	.031		.950	.179	.141	.067	.053	-.040
$\alpha = 17.6°$											
.032	-.141	.730	-.229	.204	Upper	.010	-4.284	-2.349	-2.114	-2.051	-2.046
.053	-.292	.417	-.385	.027		.080	-1.490	-2.449	-2.245	-2.157	-2.114
.100	-.196	.313	-.458	-.222		.130	-1.891	-2.290	-2.046	-1.978	-1.891
.145	-.146	.204	-.408	-.231		.145	-4.589	-2.000	-1.814	-1.772	-1.723
.189	-.077	.240	-.357	-.186		.155	-2.511	-1.869	-1.674	-1.662	-1.636
.234	-.068	.290	-.147	-.231		.180	-1.827	-1.773	-1.583	-1.561	-1.586
.280	-.096	.246	.082	-.249		.220	-1.404	-1.687	-1.510	-1.511	-1.513
.326	-.118	.299	.064	-.286		.270	-1.171	-1.551	-1.429	-1.442	-1.431
.371	-.269	.390	-.220	-.286		.400	-.843	-1.161	-1.197	-1.255	-1.221
.392	-.320	.420	-.742	-.277		.620	-.725	-.708	-.776	-.989	-.998
.413	-.387	.467	-.752	.467		.685	-1.937	-.708	-.730	-.847	-1.527
.434	-.433	.485	-1.424	.567		.693	-2.028	-.703	-.676	-.820	-1.472
.457	-.401	.500	-1.122	.608		.700	-1.426	-.644	-.612	-.755	-1.226
.480	-.387	.480	-.897	.553		.720	-.875	-.603	-.567	-.710	-.975
.502	-.428	.450	-.824	.499		.750	-.715	-.608	-.576	-.701	-.893
.551	-.346	.400	-.788	.458		.800	-.615	-.608	-.585	-.687	-.843
.585	-.314	.399	-.907	.531		.900	-.501	-.630	-.612	-.714	-.811
.592	-.310	.399	-1.003	-.898		.980	-.369	-.553	-.535	-.673	-.816
.613	-.205	.340	-.778	-.600	Lower	.025	.816	.853	.803	.778	.629
.634	-.223	.245	-.549	-.485		.120	.807	.798	.739	.705	.579
.655	-.205	.141	-.380	-.440		.220	.734	.717	.685	.655	.515
.675	-.137	.309	-.256	-.227		.300	.624	.635	.621	.586	.451
.696	-.096	.309	-.179	-.118		.620	.656	.726	.676	.604	.232
.774	-.041	.145	-.018	.059		.750	.725	.785	.694	.636	.551
.852	-.077	.077	-.037	-.036		.850	.474	.535	.458	.453	.392
.930	.018	.068	.027	.054		.950	.260	.200	.122	.092	.032

TABLE 3 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 30°$; $\delta_f = 37°$; $\delta_{a,L} = 37°$; $\delta_{a,R} = 37°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.000$ $C_{\mu,f} = 0.000$ $C_{\mu,a} = 0.000$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = 21.8°$											
.032	-.184	.780	-.373	.135	Upper	.010	-1.338	-1.361	-1.074	-1.218	-1.002
.053	-.297	.599	-.485	-.027		.080	-1.370	-1.442	-1.133	-1.281	-1.033
.100	-.171	.390	-.517	-.266		.130	-1.193	-1.392	-1.115	-1.325	-1.038
.145	-.108	.290	-.467	-.293		.145	-1.120	-1.361	-1.101	-1.308	-1.038
.189	-.018	.308	-.391	-.239		.155	-1.066	-1.274	-1.043	-1.222	-1.006
.234	.004	.367	-.184	-.275		.180	-1.025	-1.229	-1.016	-1.218	-.993
.280	-.013	.363	.045	-.266		.220	-.998	-1.197	-1.016	-1.177	-.984
.326	-.063	.372	.040	-.284		.270	-.989	-1.184	-.998	-1.195	-.975
.371	-.234	.454	-.216	-.221		.400	-.975	-1.089	-.957	-1.110	-.912
.392	-.140	.475	-.571	.131		.620	-.835	-.934	-.934	-.791	-.791
.413	-.445	.508	-.849	.501		.685	-.903	-.921	-.934	-.858	-.867
.434	-.548	.513	-1.366	.596		.693	-.839	-.985	-.776	-.800	-.800
.457	-.566	.520	-1.182	.623		.700	-.785	-.903	-.749	-.750	-.746
.480	-.539	.500	-1.132	.560		.720	-.689	-.803	-.709	-.723	-.683
.502	-.553	.460	-1.051	.515		.750	-.685	-.757	-.704	-.705	-.669
.551	-.494	.430	-.917	.451		.800	-.685	-.698	-.673	-.683	-.674
.585	-.467	.372	-.890	.506		.900	-.676	-.621	-.623	-.678	-.674
.592	-.476	.367	-.926	-.916		.980	-.612	-.522	-.564	-.652	-.656
.613	-.324	.290	-.822	-.710	Lower	.025	.803	.835	.804	.791	.634
.634	-.413	.172	-.705	-.492		.120	.825	.780	.722	.723	.593
.655	-.395	.036	-.580	-.646		.220	.739	.726	.704	.705	.544
.675	-.301	-.136	-.472	-.637		.300	.649	.644	.627	.602	.485
.696	-.261	-.150	-.373	-.537		.620	.649	.698	.664	.625	.288
.774	-.148	.073	-.112	-.099		.750	.717	.771	.718	.665	.571
.852	-.027	.145	-.058	-.126		.850	.440	.517	.447	.458	.395
.930	.045	.091	.022	.036		.950	.141	.172	.095	.094	.067

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TABLE 4

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 40°$; $\delta_f = 37°$; $\delta_{a,L} = 00°$; $\delta_{a,R} = 00°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.000$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = -1.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.268	.303	.292	.304
.053	.050	.081	.053	.058
.100	-.091	-.036	-.099	-.096
.145	-.091	-.081	-.070	-.050
.189	-.012	-.024	-.025	.004
.234	-.050	.040	.016	.004
.280	-.050	.061	.008	.017
.326	-.045	.057	-.008	.046
.371	-.095	.081		.083
.392	.008	.016	.016	.200
.413	-.132	.105	-.119	-.250
.434	-.169	.149	-.308	-.183
.457	-.194	.170	-.419	-.050
.480	-.256	.200	-.399	.108
.502	-.334	.225	-.440	.200
.551	-.367	.250	-.592	.291
.585	-.359	.275	-.711	.329
.592	-.343	.299	-.814	-.966
.613	-.252	.275	-.682	-.780
.634	-.223	.230	-.497	-.703
.655	-.186	.186	-.345	-.212
.675	-.103	.109	-.234	-.037
.696	-.050	.093	-.164	
.774	-.054	.081	-.062	.021
.852	-.021	.004	.004	-.137
.930	.066	-.194	.099	-.279

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.895	.824	.803	.801	.796
	.080	.325	.234	.175	.156	.227
	.130	-.429	-.638	-.749	-.690	-.664
	.145	-3.775	-3.454	-3.376	-3.748	-3.363
	.155	-1.374	-1.353	-1.382	-1.282	-.846
	.180	-1.091	-.877	-1.032	-.892	-.962
	.220	-.666	-.663	-.762	-.805	-.623
	.270	-.533	-.622	-.687	-.542	-.470
	.400	-.591	-.650	-.741	-.583	-.433
	.620	-1.016	-1.099	-1.016	-.399	-.235
	.685	-4.679	-4.157	-3.047		
	.693	-5.120	-5.123	-5.261		
	.700	-3.392	-3.971	-4.212	-.279	-.215
	.720	-1.657	-1.790	-1.798	-.362	-.215
	.750	-1.120	-1.135	-1.224	-.378	-.215
	.800	-.741	-.699	-.903	-.341	-.227
	.900	-.387	-.259	-.774	-.267	-.177
	.980	-.037	.121	-.350	-.148	-.103
Lower	.025	-.512	-.226	-.100	-.090	-.198
	.120	-.454	-.242	-.154	-.177	-.231
	.220	-.079	-.275	-.200	-.185	-.194
	.300	.258	-.210	-.250	-.259	-.194
	.620	.533	.558	.258	-.070	-.194
	.750	.674	.703	.083	-.177	-.144
	.850	.504	.582	.312	-.164	-.120
	.950	.387	.424	.200	-.148	-.095

$\alpha = 6.1°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.093	.468	.144	.335
.053	-.106	.238	-.070	.114
.100	-.178	.064	-.191	-.106
.145	-.131	-.013	-.165	-.064
.189	-.059	.043	-.096	-.030
.234	-.093	.106	-.035	-.038
.280	-.093	.115	.039	-.034
.326	-.097	.106	-.009	-.042
.371	-.178	.170	-.200	-.013
.392	-.210	.026	-.235	.080
.413	-.250	.255	-.487	.140
.434	-.309	.302	-.735	.322
.457	-.326	.320	-.722	.407
.480	-.369	.340	-.635	.385
.502	-.432	.360	-.640	.352
.551	-.424	.365	-.779	.390
.585	-.394	.366	-.870	.457
.592	-.369	.383	-.922	-.898
.613	-.250	.340	-.727	-.720
.634	-.237	.298	-.509	-.572
.655	-.195	.221	-.331	-.152
.675	-.110	.123	-.213	.013
.696	-.055	.098	-.139	.034
.774	-.051	.098	-.044	.034
.852	-.038	.043	-.017	-.127
.930	.059	-.128	.074	-.182

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.442	.383	.364	.470	.538
	.080	-.279	-.370	-.394	-.392	-.233
	.130	-1.271	-1.515	-1.576	-1.532	-1.411
	.145	-5.563	-5.288	-4.897	-5.508	-4.888
	.155	-2.301	-2.340	-2.275	-2.202	-1.605
	.180	-1.743	-1.566	-1.656	-1.523	-1.550
	.220	-1.082	-1.157	-1.190	-1.284	-1.038
	.270	-.841	-1.021	-1.042	-.935	-.805
	.400	-.794	-.940	-1.004	-.888	-.720
	.620	-1.107	-1.327	-1.228	-.509	-.474
	.685	-4.473	-4.544	-3.545		
	.693	-4.791	-5.701	-5.867		
	.700	-3.159	-4.318	-4.744	-.466	-.407
	.720	-1.494	-1.974	-2.135	-.557	-.424
	.750	-1.009	-1.259	-1.529	-.579	-.415
	.800	-.695	-.783	-1.195	-.500	-.394
	.900	-.369	-.294	-.974	-.418	-.326
	.980	-.047	.094	-.394	-.257	-.169
Lower	.025	-.326	.298	.059	.187	-.208
	.120	.567	.387	.419	.270	-.271
	.220	.575	.523	.572	.322	.529
	.300	.472	.494	.479	.348	.373
	.620	.601	.617	.415	.009	-.076
	.750	.725	.715	.182	-.265	-.110
	.850	.515	.553	.254	-.387	-.127
	.950	.404	.408	.161	-.278	-.148

$\alpha = 13.5°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.082	.625	-.080	.267
.053	-.260	.423	-.254	.065
.100	-.195	.207	-.356	-.164
.145	-.160	.108	-.309	-.168
.189	-.087	.147	-.258	-.129
.234	-.100	.203	-.080	-.151
.280	-.130	.211	.064	-.177
.326	-.147	.207	.042	-.220
.371	-.264	.293	-.254	-.220
.392	-.310	.340	-.589	-.164
.413	-.372	.397	-.860	.302
.434	-.416	.435	-1.114	.509
.457	-.416	.430	-1.004	.565
.480	-.455	.440	-.834	.522
.502	-.498	.450	-.775	.487
.551	-.455	.450	-.894	.487
.585	-.416	.444	-.970	.561
.592	-.381	.435	-1.063	-1.065
.613	-.264	.401	-.792	-.700
.634	-.247	.323	-.470	-.453
.655	-.204	.263	-.267	-.241
.675	-.126	.147	-.140	.034
.696	-.069	.134	-.076	.095
.774	-.048	.138	-.013	.091
.852	-.082	.056	-.047	-.073
.930	-.004	-.030	.013	-.039

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-2.274	-1.975	-2.091	-2.021	-1.784
	.080	-1.235	-1.022	-1.203	-1.216	-1.122
	.130	-2.376	-2.445	-2.613	-2.525	-2.629
	.145	-7.462	-6.731	-6.356	-6.896	-6.765
	.155	-3.282	-3.281	-3.294	-3.207	-2.638
	.180	-2.338	-2.229	-2.402	-2.232	-2.283
	.220	-1.474	-1.651	-1.755	-1.762	-1.577
	.270	-1.124	-1.384	-1.462	-1.326	-1.239
	.400	-.919	-1.134	-1.246	-1.106	-1.083
	.620	-1.085	-1.410	-1.427	-.762	-.801
	.685	-3.781	-4.286	-3.566		
	.693	-3.944	-5.299	-5.855		
	.700	-2.577	-4.057	-4.743	-.657	-.723
	.720	-1.321	-1.897	-2.255	-.733	-.758
	.750	-.902	-1.186	-1.656	-.737	-.723
	.800	-.573	-.720	-1.302	-.678	-.710
	.900	-.261	-.229	-1.035	-.517	-.598
	.980	-.051	.138	-.388	-.322	-.347
Lower	.025	.556	.694	.673	.665	.494
	.120	.774	.772	.673	.661	.511
	.220	.714	.703	.642	.597	.485
	.300	.590	.621	.569	.470	.329
	.620	.658	.690	.461	.055	-.139
	.750	.739	.789	.103	-.267	-.156
	.850	.543	.612	.315	-.407	-.182
	.950	.406	.453	.220	-.309	-.221

TABLE 4 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, F_AP, OR AILERON

Wing configuration

δ_n = 40°; δ_f = 37°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 17.2°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.142	.660	-.219	.209
.053	-.289	.448	-.367	.013
.100	-.196	.257	-.450	-.231
.145	-.156	.168	-.402	-.261
.189	-.093	.186	-.345	-.213
.234	-.093	.239	-.127	-.257
.280	-.125	.261	.070	-.283
.326	-.147	.266	.035	-.357
.371	-.303	.359	-.310	-.387
.392	-.365	.400	-.752	-.387
.413	-.436	.456	-.979	.357
.434	-.472	.496	-1.407	.570
.457	-.472	.500	-1.110	.622
.480	-.472	.490	-.905	.579
.502	-.516	.480	-.857	.535
.551	-.445	.470	-.874	.513
.585	-.392	.465	-1.075	.574
.592	-.347	.465	-1.233	-1.205
.613	-.236	.417	-.900	-.800
.634	-.214	.346	-.516	-.431
.655	-.169	.279	-.253	-.244
.675	-.089	.168	-.127	.052
.696	-.036	.146	-.048	.117
.774	-.036	.177	.009	.113
.852	-.071	.075	-.044	-.022
.930	-.009	.058	[illegible]	.048

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-6.200	-3.221	-3.381	-3.295	-3.045
	.080	-1.734	-2.915	-3.389	-3.191	-1.709
	.130	-2.775	-2.433	-2.619	-2.662	-2.911
	.145	-8.145	-6.161	-5.643	-6.189	-7.229
	.155	-3.609	-3.403	-3.350	-3.322	-3.072
	.180	-2.542	-2.450	-2.593	-2.478	-2.671
	.220	-1.629	-1.857	-1.954	-1.954	-1.896
	.270	-1.216	-1.578	-1.632	-1.538	-1.514
	.400	-.992	-1.241	-1.318	-1.219	-1.313
	.620	-1.084	-1.462	-1.462	-1.010	-1.033
	.685	-3.570	-3.704	-2.902		
	.693	-3.824	-4.676	-4.999		
	.700	-2.604	-3.594	-4.094	-.804	-.917
	.720	-1.361	-1.688	-2.006	-.861	-.953
	.750	-.922	-1.090	-1.492	-.835	-.935
	.800	-.610	-.647	-1.175	-.760	-.899
	.900	-.255	-.186	-.892	-.568	-.757
	.980	-.031	.111	-.326	-.367	-.427
Lower	.025	.716	.816	.779	.747	.601
	.120	.817	.718	.709	.691	.525
	.220	.773	.718	.700	.647	.494
	.300	.672	.611	.613	.524	.361
	.620	.681	.719	.500	.092	-.178
	.750	.751	.713	.139	-.219	-.138
	.850	.584	.6[illegible]4	.344	-.376	-.169
	.950	.439	.4[illegible]6	.244	-.337	-.214

α = 21.2°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.210	.792	-.362	.123
.053	-.335	.588	-.498	-.044
.100	-.210	.356	-.569	-.307
.145	-.192	.258	-.529	-.342
.189	-.085	.276	-.459	-.285
.234	-.094	.325	-.150	-.356
.280	-.125	.321	.097	-.382
.326	-.183	.338	.040	-.465
.371	-.349	.441	-.340	-.527
.392	-.450	.490	-.900	-.588
.413	-.505	.548	-1.125	.400
.434	-.541	.570	-1.813	.632
.457	-.479	.580	-1.323	.667
.480	-.434	.560	-1.050	.623
.502	-.434	.530	-.957	.588
.551	-.326	.500	-.922	.536
.585	-.282	.481	-1.032	.562
.592	-.264	.476	-1.261	-1.282
.613	-.148	.405	-.957	-1.000
.634	-.157	.307	-.573	-.659
.655	-.139	.240	-.304	-.167
.675	-.067	.116	-.168	.044
.696	-.022	.116	-.084	.079
.774	-.049	.196	.004	.132
.852	-.094	.098	-.053	.035
.930	.004	.093	[illegible]	.088

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-9.047	-3.8[illegible]3	-3.903	-3.762	-3.851
	.080	-1.887	-3.9[illegible]3	-4.035	-3.846	-3.627
	.130	-2.831	-2.6[illegible]4	-3.126	-3.030	-2.589
	.145	-8.019	-4.7[illegible]5	-3.864	-4.128	-6.230
	.155	-3.511	-3.0[illegible]8	-2.705	-2.589	-2.889
	.180	-2.478	-2.342	-2.230	-2.060	-2.621
	.220	-1.534	-1.798	-1.721	-1.601	-1.910
	.270	-1.105	-1.4[illegible]6	-1.427	-1.279	-1.552
	.400	-.899	-1.1[illegible]0	-1.089	-.992	-1.333
	.620	-.890	-1.1[illegible]2	-.878	-1.063	-1.082
	.685	-3.019	-2.1[illegible]7	.083		
	.693	-3.251	-2.7[illegible]6	-1.361		
	.700	-2.214	-2.1[illegible]8	-1.164	-.785	-.970
	.720	-1.181	-1.0[illegible]1	-.738	-.790	-1.024
	.750	-.814	-.674	-.694	-.776	-1.002
	.800	-.572	-.5[illegible]6	-.654	-.737	-.962
	.900	-.380	-.3[illegible]9	-.544	-.653	-.809
	.980	-.130	-.1[illegible]1	-.483	-.547	-.501
Lower	.025	.809	.8[illegible]5	.808	.776	.613
	.120	.863	.8[illegible]6	.738	.706	.514
	.220	.827	.8[illegible]1	.733	.679	.510
	.300	.720	.7[illegible]0	.650	.582	.398
	.620	.702	.7[illegible]8	.553	.198	-.161
	.750	.765	.8[illegible]0	.259	-.044	-.112
	.850	.577	.5[illegible]7	.391	-.115	-.148
	.950	.416	.3[illegible]7	.176	-.282	-.264

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TABLE 5

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = [illegible]; δ_f = [illegible]; $\delta_{a,L}$ = [illegible]; $\delta_{a,R}$ = [illegible]; h_s/c = [illegible] h_d/c = [illegible]

$C_{\mu,k}$ = [illegible] $C_{\mu,f}$ = [illegible] $C_{\mu,a}$ = [illegible]

x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)	Surface	x/c	0.221	0.426	0.640	0.800	0.918
							C_p values for spanwise stations, $\frac{y}{b/2}$, of: Wing, flap, or aileron				
α = -1.6°											
.042	.268	.322	.27	.29	Upper	.010	.844	.790	.762	.75	.783
.063	.016	.294	[illegible]	[illegible]		.080	.261	.159	.010	.018	.125
.105	-.103	-.130	-.117	.109		.130	-.513	-.738	-.362	-.935	-.850
.145	-.089	.077	-.085	-.063		.145	-4.938	3.713	-3.660	-4.499	-3.940
.189	-.013	-.021	-.065	-.024		.155	-1.462	-1.498	-1.642	-1.600	-1.069
.234	-.045	.047	.018	[illegible]		.180	-1.183	-.987	-1.252	-1.15	-1.161
.280	-.054	.069	.013	.014		.220	-.718	-.760	-.934	-1.042	-.82
.326	-.045	.082	-.074	.045		.270	-.583	-.695	-.866	-.782	-.657
.371	-.089	.112	-.121	.100		.400	-.618	-.747	-.771	-.9	-.738
.392	.018	.126	-.013	.245		.620	-1.075	-1.219	-1.348	-1.11	-1.096
.413	-.157	.155	-.157	-.082		.685	-4.799	-4.447	-3.847	-5.540	-5.801
.434	-.197	.180	-.368	-.077		.623	-5.238	-5.589	-6.504	-6.89	-5.890
.457	-.219	.200	-.485	.032		.700	-3.459	-4.232	-5.175	-4.520	-4.260
.480	-.277	.220	-.458	.145		.720	-1.688	-1.902	-2.386	-2.14	-1.767
.522	-.353	.240	-.494	.227		.750	-1.136	-1.193	-1.392	-1.45	-1.274
.551	-.376	.260	-.669	.345		.800	-.740	-.770	-.789	-.72	-1.03
.585	-.371	.296	-.791	.345		.900	-.422	-.276	-.277	-.447	-.778
.592	-.349	.309	-.903	.662		.980	-.048	.155	.136	-.03	-.44
.613	-.255	.275	-.764	-.800	Lower	.025	-.270	-.052	.077	.10	.018
.634	-.242	.240	-.566	-.748		.120	-.244	-.052	.054	.072	-.018
.655	-.174	.172	-.391	-.227		.220	-.035	-.069	.009	.040	-.016
.675	-.127	.116	-.270	-.036		.300	.248	-.026	-.005	-.013	-.081
.696	-.063	.094	-.171	.005		.620	.513	.498	.390	.297	.054
.774	-.036	.090	-.067	.032		.750	.670	.635	.449	.40	.253
.852	-.013	.026	-.013	-.141		.850	.513	.575	.508	.656	.62
.930	.067	-.355	.090	-.240		.950	.396	.451	.424	.492	[illegible]
α = 5.6°											
.042	.108	.501	.152	.336	Upper	.010	.334	.251	.100	.19	.220
.063	-.108	.264	-.067	.091		.080	-.398	-.524	-.617	-.662	-.526
.105	-.184	.087	-.197	-.104		.130	-1.470	1.727	-1.914	-1.887	-1.865
.145	-.144	.014	-.157	-.063		.145	-6.103	-5.806	-5.561	-6.29	-5.940
.189	-.258	.055	-.098	-.036		.155	-2.550	2.575	-2.644	-2.621	-2.080
.234	-.090	.114	.024	-.036		.180	-1.932	-1.745	-1.950	-1.860	-1.966
.280	-.097	.132	.254	-.032		.220	-1.218	-1.317	-1.438	-1.542	-.402
.326	-.097	.141	.336	-.036		.270	-.962	-1.162	-1.279	-1.216	-1.142
.371	-.202	.214	-.170	-.018		.400	-.875	1.066	-1.261	-1.256	-1.152
.392	-.245	.250	-.268	.062		.620	-1.220	1.482	-1.578	-1.632	-1.715
.413	-.348	.290	-.532	.200		.685	-4.628	-4.908	-4.13	-4.386	-4.603
.434	-.346	.351	-.772	.390		.693	-4.967	-6.116	-6.45	[illegible]	-6.744
.457	-.377	.360	-.760	.458		.700	-3.182	4.630	-5.479	-5.376	-6.978
.480	-.427	.370	-.777	.431		.720	-1.565	2.114	-2.372	-2.665	-3.199
.522	-.454	.380	-.689	.413		.750	-1.053	-1.331	1.524	-1.834	-2.203
.551	-.481	.390	-.809	.445		.800	-.733	-.820	-.894	1.194	-1.552
.585	-.467	.416	-.9	.517		.900	-.396	-.278	-.307	-.590	-1.143
.592	-.422	.419	-.966	-.552		.980	-.087	.132	.063	-.054	-.57
.613	-.272	.183	-.733	-.650	Lower	.025	.077	.392	.451	.438	.236
.634	-.272	.110	-.523	-.571		.120	.646	.625	.57	.644	.476
.655	-.220	.251	-.335	-.213		.220	.623	.652	.526	.635	.548
.675	-.121	.146	-.224	[illegible]		.300	.504	.570	.553	.546	.432
.696	-.076	.114	-.143	.023		.620	.627	.693	.680	.599	.018
.774	-.063	.123	-.036	.041		.750	.733	.788	.689	.644	.494
.852	-.036	.166	-.016	-.113		.850	.495	.615	.561	.537	.366
.930	.056	-.077	.063	-.122		.950	.353	.465	.41	.358	.166
α = 13.1°											
.042	-.078	.641	-.075	.268	Upper	.010	-3.513	-2.150	-2.667	-2.866	-2.567
.063	-.135	.623	-.271	.063		.080	-1.433	-1.940	-2.005	-2.348	-2.491
.105	[illegible]	.175	-.383	-.166		.130	-2.596	-2.413	-2.644	-2.717	-2.923
.145	-.156	.127	-.327	-.172		.145	-8.034	-6.686	-6.450	-7.144	-7.375
.189	-.247	.16	-.275	-.127		.155	-3.532	-3.114	-3.488	-3.517	-3.082
.234	-.115	.226	-.075	-.159		.180	-2.513	-2.310	-2.605	-2.642	-2.74
.280	-.13	.173	.065	-.17		.220	-1.627	-1.727	-1.941	-2.147	-2.00
.326	-.156	.276	.047	-.213		.270	-1.234	-1.451	-1.665	-1.708	-1.652
.371	-.281	.329	-.247	-.218		.400	-1.017	-1.211	-1.442	-1.57	-1.619
.392	-.330	.38	-.663	-.177		.620	-1.220	-1.576	-1.64	-1.84	-2.14
.413	-.391	.427	-.943	.349		.685	-4.087	-4.247	-4.452	-6.227	-6.90
.434	-.437	.463	-1.209	.549		.693	-4.285	-5.204	-5.90	-7.776	-10.20
.457	-.455	.470	-1.069	.594		.700	-2.806	-4.015	-4.722	-5.321	-8.32
.480	-.483	.480	-.882	.549		.720	-1.447	-1.874	-2.068	-2.76	-4.0
.522	-.547	.48	-.817	.508		.750	-.971	-1.189	-1.461	-1.932	-2.495
.551	-.488	.470	-.934	.531		.800	-.643	-.686	-.907	-1.126	-1.98
.585	-.465	.463	-1.083	.576		.900	-.277	-.178	-.481	-.677	-1.679
.592	-.419	.476	-1.186	-1.120		.980	-.055	.160	-.200	-.177	-.63
.613	-.271	.418	-.850	-.750	Lower	.025	.606	.752	.739	.728	.584
.634	-.262	.347	-.513	-.413		.120	.790	.753	.717	.672	.515
.655	-.225	.200	-.271	-.272		.220	.744	.729	.712	.691	.515
.675	-.124	.169	-.149	.014		.300	.629	.654	.653	.588	.423
.696	-.078	.142	-.075	.086		.620	.686	.739	.594	.607	-.097
.774	-.037	.187	.005	.104		.750	.767	.852	.667	.625	.437
.852	-.060	.076	-.028	-.054		.850	.573	.550	.553	.527	.454
.930	.014	.022	.019	.027		.950	.425	.494	.163	.359	.175

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TABLE 5 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 40°$; $\delta_f = 37°$; $\delta_{a,L} = 37°$; $\delta_{a,R} = 37°$; $h_s/c = 1.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = 16.9°$

x/l	Fuselage, 0.000, Upper surface	Fuselage, 0.000, Lower surface	Fuselage, 0.154, Upper surface	Fuselage, 0.154, Lower surface
.032	-.138	.733	-.243	.184
.053	-.299	.518	-.397	-.005
.100	-.189	.266	-.476	-.248
.145	-.170	.191	-.425	-.271
.189	-.097	.219	-.373	-.216
.234	-.101	.261	-.107	-.262
.280	-.129	.261	.093	-.290
.326	-.166	.299	.070	-.345
.371	-.317	.392	-.303	-.405
.392	-.390	.450	-.840	-.414
.413	-.460	.509	-1.074	.400
.434	-.520	.518	-1.568	.603
.457	-.488	.520	-1.214	.639
.480	-.497	.520	-.990	.612
.502	-.534	.520	-.934	.575
.551	-.469	.500	-.971	.557
.585	-.432	.495	-1.181	.603
.592	-.386	.495	-1.284	-1.215
.613	-.244	.453	-.934	-.850
.634	-.235	.373	-.541	-.492
.655	-.175	.303	-.266	-.271
.675	-.106	.173	-.107	.083
.696	-.060	.154	-.037	.124
.774	-.047	.221	.042	.133
.852	-.064	.093	-.037	-.014
.930	.009	.089	.009	.064

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-7.075	-3.575	-3.892	-3.968	-3.823
	.080	-1.812	-3.355	-4.058	-4.038	-3.450
	.130	-2.866	-2.437	-2.802	-3.207	-2.862
	.145	-8.463	-6.174	-5.507	-5.919	-7.241
	.155	-3.744	-3.531	-3.427	-3.543	-3.349
	.180	-2.657	-2.537	-2.756	-2.801	-3.036
	.220	-1.705	-1.555	-2.116	-2.268	-2.264
	.270	-1.263	-1.615	-1.803	-1.876	-1.928
	.400	-1.013	-1.326	-1.486	-1.634	-1.840
	.620	-1.115	-1.545	-1.573	-2.353	-2.314
	.685	-3.563	-3.733	-2.420	-4.532	-10.623
	.693	-3.758	-4.626	-4.605	-5.634	-10.982
	.700	-2.508	-3.603	-3.681	-3.827	-8.976
	.720	-1.338	-1.699	-1.601	-2.082	-4.380
	.750	-.910	-1.064	-1.077	-1.526	-3.225
	.800	-.585	-.611	-.768	-1.055	-2.461
	.900	-.237	-.159	-.584	-.635	-1.504
	.980	-.037	.140	-.437	-.369	-.511
Lower	.025	.748	.840	.801	.789	.603
	.120	.836	.831	.732	.709	.501
	.220	.785	.734	.732	.709	.520
	.300	.678	.719	.667	.621	.442
	.620	.706	.755	.704	.616	-.097
	.750	.776	.836	.685	.630	.432
	.850	.599	.653	.534	.527	.386
	.950	.446	.504	.308	.265	.244

$\alpha = 21.2°$

x/l	Fuselage, 0.000, Upper surface	Fuselage, 0.000, Lower surface	Fuselage, 0.154, Upper surface	Fuselage, 0.154, Lower surface
.032	-.198	.768	-.385	.091
.053	-.297	.557	-.505	-.082
.100	-.180	.363	-.555	-.333
.145	-.162	.258	-.519	-.369
.189	-.081	.281	-.452	-.314
.234	-.054	.313	-.134	-.374
.280	-.099	.331	.098	-.401
.326	-.153	.350	.076	-.492
.371	-.341	.451	-.344	-.556
.392	-.420	.490	-.886	-.611
.413	-.503	.534	-1.118	.410
.434	-.521	.561	-1.825	.620
.457	-.472	.560	-1.324	.670
.480	-.441	.540	-1.087	.615
.502	-.431	.520	-.962	.592
.551	-.337	.500	-.948	.524
.585	-.297	.478	-1.060	.556
.592	-.279	.478	-1.342	-1.358
.613	-.153	.386	-1.047	-1.000
.634	-.166	.327	-.680	-.702
.655	-.139	.225	-.398	-.251
.675	-.081	.110	-.246	-.005
.696	-.045	.101	-.139	.036
.774	-.031	.175	[illegible]	.109
.852	-.085	.097	-.058	.018
.930	[illegible]	.101	-.204	.091

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-8.672	-3.501	-3.778	-3.439	-3.554
	.080	-1.852	-4.016	-3.869	-3.520	-3.289
	.130	-2.768	-2.747	-3.231	-2.943	-2.112
	.145	-7.916	-4.638	-3.135	-3.050	-4.992
	.155	-3.439	-2.570	-2.320	-2.021	-2.184
	.180	-2.415	-2.319	-1.914	-1.677	-2.103
	.220	-1.503	-1.735	-1.527	-1.337	-1.501
	.270	-1.087	-1.458	-1.353	-1.163	-1.236
	.400	-.890	-1.159	-1.226	-1.042	-1.357
	.620	-.921	-1.072	-1.048	-1.100	-1.173
	.685	-3.086	-1.656	.164	-1.449	-2.651
	.693	-3.336	-2.078	-1.476	-1.834	-2.399
	.700	-2.276	-1.753	-1.322	-1.266	-2.004
	.720	-1.225	-.925	-.893	-.872	-1.150
	.750	-.863	-.736	-.848	-.863	-1.060
	.800	-.608	-.755	-.811	-.796	-.993
	.900	-.371	-.695	-.738	-.707	-.921
	.980	-.152	-.534	-.665	-.666	-.840
Lower	.025	.814	.870	.820	.801	.647
	.120	.854	.851	.752	.729	.584
	.220	.787	.801	.756	.738	.602
	.300	.707	.722	.674	.662	.535
	.620	.693	.745	.684	.672	.225
	.750	.760	.619	.674	.631	.576
	.850	.555	.598	.478	.474	.386
	.950	.398	.252	.109	.139	.063

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TABLE 6

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 37°; $\delta_{a,L}$ = 37°; $\delta_{a,R}$ = 37°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					α = -1.4°						
.032	.253	.299	.271	.278	Upper	.010	.936	.852	.818	.837	.829
.053	.033	.080	.071	.046		.080	.443	.358	.278	.246	.298
.100	-.091	-.034	-.092	-.093		.130	-.360	-.599	-.763	-.724	-.692
.145	-.066	-.072	-.058	-.055		.145	-4.431	-4.272	-4.171	-4.645	-4.207
.189	-.017	-.021	.008	-.013		.155	-1.656	-1.704	-1.788	-1.727	-1.227
.234	-.037	.046	.017	.004		.180	-1.258	-1.105	-1.303	-1.161	-1.206
.280	-.041	.072	.037	.021		.220	-.753	-.822	-.940	-1.011	-.800
.326	-.037	.076	.012	.059		.270	-.602	-.751	-.852	-.749	-.630
.371	-.020	.110	-.087	.122		.400	-.640	-.768	-.924	-.837	-.663
.392	.017	.120	.021	.120		.620	-1.028	-1.219	-1.286	-.853	-.912
.413	-.133	.131	-.129	-.072		.685	-4.624	-4.415	-3.627	-4.866	-3.908
.434	-.174	.143	-.362	-.236		.693	-5.033	-5.571	-5.993	-6.173	-3.631
.457	-.207	.017	-.487	-.169		.700	-3.298	-4.192	-4.787	-3.959	-2.752
.480	-.261	.017	-.433	-.055		.720	-1.580	-1.877	-2.020	-1.844	-1.107
.502	-.348	.017	-.458	.059		.750	-1.012	-1.172	-1.261	-1.224	-.862
.551	-.365	.017	-.620	.245		.800	-.669	-.700	-.709	-.720	-.758
.585	-.361	.224	-.716	.240		.900	-.431	-.202	-.245	-.337	-.630
.592	-.352	.261	-.820	-.784		.980	-.046	.194	.122	-.042	-.477
.613	-.253	.245	-.666	-.710	Lower	.025	-.184	-.143	.038	.096	-.025
.634	-.216	.219	-.470	-.658		.120	-.268	-.110	.025	.037	-.050
.655	-.170	.177	-.316	-.409		.220	-.226	-.122	.000	.012	-.050
.675	-.087	.105	-.208	-.051		.300	-.084	-.156	-.055	-.017	-.075
.696	-.041	.105	-.133	.004		.620	.401	.287	.089	-.021	-.224
.774	-.058	.089	-.037	.017		.750	.627	.481	.169	.129	.108
.852	-.037	.017	-.008	-.148		.850	.598	.590	.367	.254	.228
.933	.062	-.127	.083	-.202		.950	.443	.506	.396	.304	.157
					α = 5.7°						
.032	.088	.459	.138	.326	Upper	.010	.610	.589	.518	.556	.573
.053	-.106	.234	-.071	.083		.080	-.158	-.225	-.322	-.365	-.247
.100	-.190	.065	-.187	-.109		.130	-1.344	-1.555	-1.775	-1.741	-1.738
.145	-.150	[illegible]	-.160	-.083		.145	-6.748	-6.419	-6.270	-7.069	-6.625
.189	-.071	.043	-.111	-.030		.155	-2.806	-2.807	-2.954	-2.956	-2.369
.234	-.097	.117	-.036	-.048		.180	-2.046	-1.819	-2.093	-2.003	-2.091
.280	-.097	.126	.067	-.030		.220	-1.291	-1.347	-1.518	-1.674	-1.469
.326	-.115	.130	.018	-.030		.270	-.992	-1.165	-1.327	-1.291	-1.195
.371	-.194	.191	-.178	.013		.400	-.891	-1.048	-1.270	-1.287	-1.235
.392	-.230	.220	-.258	.170		.620	-1.185	-1.416	-1.588	-1.585	-1.658
.413	-.287	.269	-.543	.152		.685	-4.193	-4.656	-4.203	-6.397	-7.970
.434	-.353	.312	-.837	.165		.693	-4.404	-5.787	-6.674	-8.004	-8.045
.457	-.371	.320	-.810	.309		.700	-2.849	-4.375	-5.352	-5.377	-6.404
.480	-.423	.330	-.721	.392		.720	-1.405	-1.997	-2.354	-2.680	-3.013
.502	-.498	.350	-.699	.409		.750	-.979	-1.269	-1.501	-1.847	-2.223
.551	-.481	.370	-.788	.439		.800	-.724	-.767	-.879	-1.224	-1.786
.585	-.450	.390	-.886	.448		.900	-.413	-.247	-.352	-.592	-1.266
.592	-.423	.381	-.926	-.875		.980	-.162	.152	.070	-.062	-.538
.613	-.300	.347	-.712	-.600	Lower	.025	.079	.351	.339	.365	.137
.634	-.273	.295	-.490	-.474		.120	.136	.303	.287	.249	.031
.655	-.221	.225	-.329	-.387		.220	.487	.347	.300	.267	.137
.675	-.132	.121	-.196	-.117		.300	.584	.472	.500	.485	.450
.696	-.088	.095	-.116	-.009		.620	.623	.645	.631	.614	.141
.774	-.057	.126	-.027	.026		.750	.724	.710	.666	.641	.512
.852	-.057	.052	-.022	-.157		.850	.518	.593	.583	.548	.406
.930	.040	-.065	.045	-.100		.950	.360	.450	.435	.383	.194
					α = 9.4°						
.032	.000	.567	-.161	.276	Upper	.010	.358	.222	.053	.087	.140
.053	-.184	.340	-.274	.058		.080	-.487	-.608	-.770	-.805	-.669
.100	-.179	.113	-.226	-.147		.130	-1.798	-2.200	-2.448	-2.354	-2.351
.145	-.149	.045	-.187	-.134		.145	-7.701	-7.851	-7.541	-8.210	-7.745
.189	-.074	.100	-.074	-.089		.155	-3.242	-3.529	-3.624	-3.576	-2.950
.234	-.105	.159	.074	-.107		.180	-2.312	-2.286	-2.546	-2.415	-2.483
.280	-.109	.172	.017	-.107		.220	-1.480	-1.660	-1.843	-1.932	-1.731
.326	-.131	.168	-.204	-.134		.270	-1.105	-1.406	-1.580	-1.501	-1.403
.371	-.249	.245	-.435	-.111		.400	-.935	-1.193	-1.438	-1.423	-1.407
.392	-.290	.068	-.748	-.036		.620	-1.011	-1.483	-1.700	-1.853	-1.892
.413	-.328	.349	-1.031	.142		.685	-2.715	-4.722	-4.371	-6.592	-9.117
.434	-.389	.404	-.931	.423		.693	-2.657	-5.864	-6.873	-8.227	-9.344
.457	-.411	.410	-.783	.521		.700	-1.623	-4.422	-5.524	-5.573	-7.487
.480	-.446	.420	-.735	.494		.720	-.796	-1.996	-2.439	-2.841	-3.558
.502	-.511	.430	-.822	.454		.750	-.622	-1.225	-1.563	-1.954	-2.649
.551	-.463	.440	-.801	.450		.800	-.564	-.730	-.935	-1.314	-2.128
.585	-.428	.431	-.753	.525		.900	-.456	-.236	-.365	-.653	-1.534
.592	-.402	.449	-.574	-.766		.980	-.335	.150	.071	-.074	-.638
.613	-.306	.381	-.431	-.450	Lower	.025	.063	.531	.512	.548	.367
.634	-.271	.313	-.309	-.338		.120	.666	.658	.694	.666	.516
.655	-.236	.222	-.183	-.383		.220	.711	.726	.708	.705	.577
.675	-.135	.122	-.091	-.236		.300	.604	.649	.619	.596	.446
.696	-.092	.095	.039	-.076		.620	.648	.712	.694	.618	-.057
.774	-.039	.154	-.039	.018		.750	.720	.803	.694	.631	.455
.852	-.079	.073	.013	-.129		.850	.505	.635	.579	.526	.350
.930	.004	-.018	-.004	-.009		.950	.295	.472	.432	.379	.162

TABLE 5 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 37°; $\delta_{a,L}$ = 37°; $\delta_{a,R}$ = 47°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.104

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 13.1°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.425	0.640	0.800	0.918
.032	-.064	.552	-.101	.275	Upper	.010	-.197	-1.376	-1.675	-1.895	-1.486
.053	-.226	.427	-.276	.059		.080	-1.021	-1.102	-1.232	-1.403	-1.192
.100	-.185	.202	-.363	-.176		.130	-2.532	-2.852	-3.133	-3.156	-3.196
.145	-.135	.117	-.427	-.163		.145	-8.906	-9.075	-8.600	-9.666	-9.182
.189	-.081	.155	-.171	-.112		.155	-3.800	-4.198	-4.325	-4.472	-3.720
.234	-.104	.216	-.110	.172		.180	-2.596	-2.737	-3.020	-3.013	-3.070
.280	-.126	.211	.087	-.176		.220	-1.676	-1.975	-2.171	-2.333	-2.126
.326	-.143	.220	.060	-.230		.270	-1.268	-1.647	-1.810	1.826	1.729
.371	-.271	.324	-.244	-.230		.400	-1.071	-1.327	-1.557	1.638	-1.679
.392	-.322	.370	-.644	-.203		.620	-1.209	-1.557	-1.801	-2.259	-2.239
.413	-.388	.427	-.980	.194		.685	-3.471	-4.715	-4.379	-7.8[illegible]2	-10.220
.434	-.438	.464	-1.288	.519		.693	-3.512	-5.853	-6.871	-8.654	-10.541
.457	-.442	.465	-1.090	.600		.700	-2.459	-4.415	-5.530	-5.783	-8.523
.481	-.474	.468	-.971	.555		.720	-1.369	-2.017	-2.485	-3.050	-4.171
.502	-.533	.470	-.934	.510		.750	-.966	-1.252	-1.605	-2.121	-3.137
.551	-.479	.470	-1.012	.515		.800	-.627	-.713	-.943	-1.389	-2.537
.585	-.451	.474	-1.063	.569		.900	-.256	-.167	-.361	-.658	-1.783
.592	-.405	.478	-1.113	-1.061		.980	-.041	.207	.072	-.064	-.682
.613	-.289	.427	-.842	-.700	Lower	.025	.472	.694	.673	.714	.537
.634	-.253	.361	-.488	-.429		.120	.810	.816	.736	.709	.555
.655	-.212	.267	-.271	.235		.220	.778	.788	.740	.713	.569
.675	-.122	.159	-.147	.036		.300	.559	.704	.668	.644	.460
.696	-.081	.150	-.087	.099		.620	.591	.755	.709	.640	-.108
.774	-.041	.178	.005	.086		.750	.765	.631	.691	.639	.442
.852	-.072	.140	-.051	-.050		.850	.563	.665	.587	.557	.352
.930	.018	.043	.014	.014		.920	.395	.501	.460	.409	.199

α = 16.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.425	0.640	0.800	0.918
.032	-.153	.714	-.240	.187	Upper	.010	-2.379	-2.441	-2.503	-3.257	-2.813
.053	-.297	.512	-.390	.000		.080	-1.564	-1.277	-1.964	-2.555	-1.573
.100	-.207	.270	-.454	.242		.130	-3.087	-3.137	-3.413	-3.420	-3.693
.145	-.162	.189	-.431	-.260		.145	-9.856	-9.413	-9.046	-9.888	-9.993
.189	-.099	.220	-.372	-.219		.155	-4.251	-4.487	-4.717	-4.876	-4.260
.234	-.103	.265	-.154	-.269		.180	-2.862	-2.971	-3.140	-3.370	-3.487
.280	-.130	.274	.104	-.287		.220	-1.799	-2.125	-2.424	-2.608	-2.480
.326	-.157	.292	.059	-.355		.270	-1.320	-1.734	-1.991	-2.059	-2.031
.371	-.315	.382	-.292	-.406		.400	-1.040	-1.363	-1.654	1.764	-1.941
.392	-.375	.430	-.821	-.451		.620	-1.058	-1.559	-1.850	2.540	2.498
.413	-.449	.494	-1.111	.191		.685	-3.720	-4.126	-4.083	-6.416	-11.184
.434	-.503	.535	-1.519	.588		.693	-3.395	-5.100	-6.530	-8.078	-11.624
.457	-.485	.540	-1.377	.661		.700	-2.791	-3.864	-5.254	-5.470	-9.490
.481	-.494	.535	-1.125	.629		.720	-1.701	-1.773	[illegible]	-2.975	-4.758
.502	-.540	.520	-1.043	.588		.750	-.828	-1.073	1.545	2.104	3.566
.551	-.449	.510	-1.157	.542		.800	-.524	-.584	-.502	[illegible]	2.535
.585	-.404	[illegible]	-1.270	.626		.900	-.317	-.08[illegible]	[illegible]	-.606	1.820
.592	-.368	.499	-1.216	-.100		.980	-.189	.225	.041	.077	-.529
.613	-.270	.427	-.789	-.600	Lower	.025	.644	.791	.793	.776	.593
.634	-.234	.364	-.444	-.342		.120	.879	.827	.770	.712	.494
.655	-.189	.283	-.227	-.305		.220	.828	.811	.802	.739	.548
.675	-.104	.166	-.113	-.050		.300	.736	.721	.711	.671	.458
.696	-.054	.155	-.041	.064		.620	.727	.761	.757	.645	-.157
.774	-.004	.207	.036	.123		.750	.787	.845	.734	.667	.404
.852	-.076	.099	-.046	-.073		.850	.584	.665	.625	.571	.346
.930	.004	.085	-.018	.073		.950	.409	.501	.465	.421	.220

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TABLE

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
α = -1.6°				
.032	.254	.296	.287	.295
.053	.047	.074	.051	.052
.100	-.108	-.039	-.110	-.091
.145	-.103	-.078	-.076	-.052
.189	-.030	-.035	[illegible]	-.029
.234	-.069	.035	-.030	.017
.280	-.065	.061	-.072	.043
.326	-.047	.065	-.042	.074
.371	-.121	.096	-.063	.143
.392	-.060	-.061	-.008	.325
.413	-.168	.117	-.143	-.368
.434	-.203	.144	-.367	-.368
.457	-.237	-.065	-.477	-.208
.480	-.306	-.065	-.464	-.039
.502	-.388	-.065	-.510	.303
.551	-.423	-.065	-.742	.416
.585	-.427	.313	-.894	.403
.592	-.401	.339	-1.071	-.962
.613	-.323	.278	-.894	-.806
.634	-.259	.183	-.641	-.589
.655	-.203	.113	-.451	-.156
.675	-.125	.030	-.312	-.056
.696	-.060	.004	-.215	-.009
.774	-.078	.074	-.101	.017
.852	-.095	-.078	-.105	-.121
.930	.022	-.017	.013	.017
α = 9.5°				
.032	.013	.557	.035	.321
.053	-.152	.337	-.150	.116
.100	-.191	.130	-.265	-.102
.145	-.148	.058	-.229	-.089
.189	-.074	.085	-.163	-.045
.234	-.096	.157	-.115	-.071
.280	-.100	.166	.035	-.062
.326	-.113	.166	.018	-.098
.371	-.213	.247	-.203	-.080
.392	-.270	.280	-.432	.022
.413	-.326	.328	-.741	.169
.434	-.387	.391	-1.014	.436
.457	-.405	.400	-.887	.543
.480	-.439	.410	-.776	.525
.502	-.513	.420	-.741	.499
.551	-.479	.435	-.856	.534
.585	-.435	.445	-.829	.614
.592	-.413	.436	-.790	-.779
.613	-.322	.355	-.618	-.600
.634	-.283	.247	-.485	-.485
.655	-.231	.139	-.371	-.441
.675	-.148	.031	-.225	-.196
.696	-.091	.031	-.106	-.009
.774	.039	.126	.018	.062
.852	-.078	.022	-.049	-.102
.930	.013	[illegible]	.018	.027
α = 13.4°				
.032	-.067	.632	-.092	.274
.053	-.225	.420	-.273	.063
.100	-.184	.199	-.370	-.171
.145	-.135	.122	-.324	-.180
.189	-.090	.153	-.273	-.126
.234	-.090	.208	-.185	-.157
.280	-.108	.208	.046	-.180
.326	-.130	.208	.046	-.216
.371	-.252	.293	-.236	-.234
.392	-.300	.350	-.629	-.175
.413	-.364	.402	-.934	.189
.434	-.409	.447	-1.211	.521
.457	-.418	.450	-1.036	.607
.480	-.449	.455	-.920	.575
.502	-.503	.465	-.878	.535
.551	-.449	.470	-.925	.571
.585	-.422	.465	-.901	.643
.592	-.400	.456	-.906	-.818
.613	-.297	.375	-.740	-.600
.634	-.265	.248	-.541	-.404
.655	-.220	.135	-.342	-.422
.675	-.135	-.009	-.203	-.261
.696	-.076	.027	-.116	-.072
.774	-.077	.140	.032	.081
.852	-.081	.045	-.051	-.058
.930	[illegible]	.027	[illegible]	.031

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
α = -1.6°						
Upper	.010	.962	.901	.866	.852	.832
	.080	.479	.357	.316	.316	.349
	.130	-.339	-.622	-.684	-.620	-.582
	.145	-4.490	-4.407	-4.071	-4.462	-3.894
	.155	-1.688	-1.753	-1.702	-1.590	-1.117
	.180	-1.292	-1.144	-1.234	-1.075	-1.121
	.220	-.792	-.853	-.879	-.924	-.737
	.270	-.635	-.779	-.983	-.662	-.556
	.400	-.700	-.827	-.866	-.692	-.509
	.620	-1.310	-1.497	-1.269	-.308	-.328
	.665	-6.792	-6.844	-5.293		
	.693	-6.922	-7.475	-7.164		
	.700	-4.407	-5.426	-5.548	-.371	-.375
	.720	-2.149	-2.419	-2.443	-.472	-.323
	.750	1.444	-1.510	-1.741	-.477	-.315
	.800	-.992	-.914	-1.386	-.455	-.306
	.900	-.553	-.331	-1.213	-.312	-.272
	.980	-.013	.148	-.472	-.177	-.220
Lower	.025	-.553	-.335		.059	-.177
	.120	-.431	-.300	-.065	-.042	-.204
	.220	-.331	-.374	-.074	-.076	-.177
	.300	-.004	-.513	-.139	-.181	-.216
	.620	.679	.722	.251	-.434	-.336
	.750	.831	.870	.082	-.287	-.272
	.850	.679	.757	.359	-.164	-.250
	.950	.544	.583	.286	-.160	-.216
α = 9.5°						
Upper	.010	.376	.274	.240	.348	.457
	.080	-.450	-.571	-.614	-.591	-.396
	.130	-1.718	-2.062	-2.212	-2.095	-1.910
	.145	-7.500	-7.544	-7.149	-7.644	-6.674
	.155	-3.147	-3.361	-3.401	-3.266	-2.510
	.180	-2.238	-2.161	-2.364	-2.157	-2.088
	.220	-1.416	-1.573	-1.678	-1.707	-1.418
	.270	-1.058	-1.325	-1.625	-1.266	-1.086
	.400	-.896	1.132	-1.273	-1.107	-.940
	.620	-1.018	-1.591	-1.594	-.626	-.648
	.685	-2.941	-6.493	-6.108		
	.693	-2.596	-7.077	-8.093		
	.700	-1.416	-5.082	-6.295	-.626	-.666
	.720	-.682	-2.197	-2.929	-.723	-.613
	.750	-.594	-1.312	-2.155	-.741	-.579
	.800	-.559	-.768	-1.772	-.688	-.561
	.900	-.472	-.283	-1.478	-.543	-.483
	.980	-.398	.139	-.525	-.366	-.305
Lower	.025	.013	.490	.445	.379	-.030
	.120	.669	.598	.623	.459	.265
	.220	.721	.692	.739	.631	.553
	.300	.629	.625	.592	.494	.400
	.620	.726	.741	.494	-.026	-.122
	.750	.822	.818	.027	-.415	-.096
	.850	.634	.687	.418	-.534	-.161
	.950	.385	.526	.312	-.415	-.200
α = 13.4°						
Upper	.010	-.075	-1.201	-1.361	-1.036	-.112
	.080	-.917	-1.011	-1.146	-1.211	-.854
	.130	-2.331	-2.681	-2.961	-2.820	-2.615
	.145	-8.632	-8.636	-8.339	-9.051	-8.052
	.155	-3.664	-3.936	-4.116	-4.036	-3.136
	.180	-2.495	-2.555	-2.822	-2.649	-2.534
	.220	-1.455	-1.847	-2.013	-2.020	-1.716
	.270	-1.165	-1.517	-1.847	-1.535	-1.303
	.400	-.970	-1.223	-1.406	-1.267	-1.119
	.620	-1.019	-1.589	-1.689	-.809	-.840
	.685	-2.743	-5.103	-6.147		
	.693	-2.497	-6.627	-8.101		
	.700	-1.422	-4.704	-6.331	-.744	-.746
	.720	-.651	-2.022	-2.092	-.878	-.777
	.750	-.549	-1.174	-2.211	-.860	-.768
	.800	-.510	-.646	-1.802	-.809	-.746
	.900	-.443	-.226	-1.483	-.619	-.643
	.980	-.346	.144	-.499	-.425	-.177
Lower	.025	.448	.668	.647	.596	.409
	.120	.775	.785	.719	.689	.589
	.220	.758	.767	.719	.656	.535
	.300	.660	.668	.607	.509	.368
	.620	.744	.750	.512	-.018	-.144
	.750	.837	.880	.184	-.453	-.171
	.850	.607	.718	.440	-.573	-.202
	.950	.386	.542	.415	-.462	-.252

TABLE 7 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 17.2°											
.032	-.132	.729	-.225	.201	Upper	.010	-1.759	-2.323	-2.474	-2.494	-2.055
.053	-.281	.539	-.377	.009		.080	-1.476	-1.273	-1.604	-1.523	-1.315
.100	-.186	.297	-.446	-.246		.130	-2.930	-3.089	-3.272	-3.322	-3.356
.145	-.145	.195	-.405	-.255		.145	-9.529	-9.262	-8.627	-9.726	-9.067
.189	-.086	.228	-.359	-.210		.155	-4.120	-4.422	-4.448	-4.555	-3.733
.234	-.082	.269	-.235	-.260		.180	-2.775	-2.899	-3.090	-3.032	-2.975
.280	-.118	.288	.074	-.283		.220	-1.713	-2.081	-2.206	-2.264	-2.023
.326	-.154	.283	.074	-.355		.270	-1.240	-1.668	-1.928	-1.739	-1.574
.371	-.290	.390	-.276	-.387		.400	-.975	-1.310	-1.449	-1.371	-1.343
.392	-.350	.440	-.805	-.419		.620	-.952	-1.607	-1.704	-1.035	-1.034
.413	-.422	.502	-1.031	.173		.685	-2.930	-5.342	-5.623		
.434	-.467	.530	-1.449	.570		.693	-2.730	-5.792	-7.519		
.457	-.472	.535	-1.371	.656		.700	-1.636	-4.181	-5.888	-.865	-.925
.480	-.472	.540	-1.063	.611		.720	-.725	-1.802	-2.816	-.943	-.971
.502	-.517	.540	-.975	.579		.750	-.524	-1.003	-2.064	-.934	-.943
.551	-.440	.525	-1.031	.556		.800	-.483	-.516	-1.682	-.860	-.921
.585	-.399	.511	-1.187	.633		.900	-.383	-.135	-1.303	-.662	-.780
.592	-.358	.506	-1.118	-1.039		.980	-.324	.130	-.374	-.442	-.472
.613	-.272	.413	-.759	-.700	Lower	.025	.583	.799	.747	.709	.526
.634	-.245	.288	-.478	-.387		.120	.829	.836	.711	.676	.549
.655	-.200	.167	-.281	-.346		.220	.793	.808	.738	.672	.513
.675	-.113	.065	-.161	-.196		.300	.706	.729	.620	.570	.381
.696	-.068	.065	-.078	-.036		.620	.761	.808	.510	.037	-.222
.774	-.063	.195	.051	.109		.750	.834	.864	.264	-.405	-.168
.852	-.059	.065	-.023	-.041		.850	.652	.739	.433	-.524	-.209
.930	.027	.074	.027	.036		.950	.424	.567	.337	-.428	-.277

L-9?7

TABLE 8

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = -1.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.272	.324	.285	.298
.053	.053	.082	.053	.050
.100	-.080	-.050	-.085	-.090
.145	-.080	-.082	-.062	-.054
.189	-.009	-.027	.000	.000
.234	-.045	.055	-.027	.018
.280	-.040	.087	-.009	.036
.326	-.036	.082	-.009	.090
.371	-.116	.141	-.093	.153
.392	-.018	-.018	-.027	.384
.413	-.187	.182	-.214	.149
.434	-.231	.205	-.463	.009
.457	-.267	-.023	-.552	.009
.480	-.338	-.027	-.530	.095
.502	-.418	-.027	-.579	.167
.551	-.454	-.027	-.766	.307
.585	-.436	.287	-.904	.226
.592	-.418	.287	-1.033	-.939
.613	-.325	.242	-.837	-.664
.634	-.258	.159	-.597	-.948
.655	-.200	.109	-.396	-.199
.675	-.120	.046	-.249	-.072
.696	-.062	.041	-.165	-.041
.774	.142	.087	-.053	.000
.852	-.089	-.027	-.080	-.153
.930	.000	-.050	.000	-.009

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.900	.834	.804	.797	.801
	.080	.342	.214	.144	.125	.218
	.130	-.496	-.843	-.975	-.930	-.899
	.145	-4.878	-4.954	-4.772	-5.284	-4.772
	.155	-1.879	-2.037	-2.077	-2.012	-1.527
	.180	-1.436	-1.349	-1.539	-1.389	-1.451
	.220	-.869	-1.012	-1.124	-1.215	-1.002
	.270	-.702	-.921	-1.228	-.917	-.801
	.400	-.764	-.971	-1.147	-1.059	-.873
	.620	-1.256	-1.618	-1.679	-1.304	-1.340
	.685	-6.437	-6.986	-6.419	-8.066	-6.904
	.693	-6.314	-7.656	-8.320	-8.957	-5.769
	.700	-3.947	-5.519	-6.320	-5.529	-4.425
	.720	-1.822	-2.429	-2.654	-2.631	-1.923
	.750	-1.155	-1.454	-1.643	-1.683	-1.478
	.800	-.856	-.834	-.876	-.975	-1.197
	.900	-.637	-.214	-.266	-.436	-.957
	.980	-.088	.273	.190	.018	-.668
Lower	.025	.066	.132	.190	.205	.045
	.120	-.009	.128	.167	.160	.027
	.220	-.013	.100	.140	.125	.022
	.300	.097	.068	.090	.062	-.067
	.620	.294	.278	.221	.205	.053
	.750	.474	.442	.357	.343	.387
	.850	.672	.583	.515	.499	.499
	.950	.580	.611	.551	.481	.321

α = 9.0°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.005	.561	[illegible]	.299
.053	-.159	.316	-.160	.061
.100	-.178	.132	-.279	-.145
.145	-.123	.057	-.234	-.126
.189	-.064	.104	-.179	-.070
.234	-.096	.156	-.119	-.107
.280	-.114	.165	.060	-.107
.326	-.128	.189	.027	-.135
.371	-.237	.278	-.169	-.131
.392	-.300	.320	-.522	-.075
.413	-.360	.372	-.847	.182
.434	-.428	.429	-1.122	.481
.457	-.447	.440	-.984	.583
.480	-.492	.460	-.884	.537
.502	-.565	.480	-.847	.509
.551	-.524	.478	-.911	.555
.585	-.492	.476	-.884	.611
.592	-.456	.471	-.838	-.742
.613	-.365	.377	-.664	-.510
.634	-.301	.259	-.527	-.481
.655	-.260	.137	-.398	-.481
.675	-.155	.033	-.252	-.233
.696	-.114	.024	-.124	-.037
.774	-.090	.132	.041	.033
.852	-.064	.047	-.041	-.126
.930	.018	.005	.018	-.014

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.183	-.028	-.378	-.746	-.064
	.080	-.689	-.811	-.985	-1.090	-.820
	.130	-2.115	-2.451	-2.740	-2.692	-2.652
	.145	-8.494	-8.347	-8.126	-9.006	-8.330
	.155	-3.588	-3.790	-3.991	-4.043	-3.313
	.180	-2.514	-2.475	-2.810	-2.697	-2.725
	.220	-1.599	-1.819	-2.054	-2.166	-1.937
	.270	-1.215	-1.532	-1.956	-1.699	-1.586
	.400	-1.037	-1.315	-1.648	-1.667	-1.631
	.620	-1.126	-1.768	-2.105	-2.257	-2.397
	.685	-2.861	-6.773	-7.230	-10.197	-14.054
	.693	-2.491	-7.400	-9.163	-11.342	-13.124
	.700	-1.393	-5.274	-7.015	-7.221	-10.185
	.720	-.708	-2.281	-3.067	-3.658	-4.876
	.750	-.633	-1.343	-1.904	-2.445	-3.591
	.800	-.568	-.764	-1.092	-1.557	-2.853
	.900	-.483	-.231	-.378	-.710	-1.937
	.980	-.417	.146	.098	-.018	-.674
Lower	.025	.319	.594	.611	.641	.460
	.120	.755	.778	.723	.737	.570
	.220	.746	.768	.756	.719	.579
	.300	.643	.683	.677	.627	.447
	.620	.755	.778	.756	.682	-.128
	.750	.830	.872	.747	.678	.483
	.850	.605	.735	.695	.650	.419
	.950	.338	.547	.537	.499	.255

α = 18.7°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.154	.741	-.325	.163
.053	-.294	.562	-.469	-.009
.100	-.191	.332	-.534	-.271
.145	-.159	.247	-.497	-.308
.189	-.065	.256	-.446	-.257
.234	-.075	.301	-.288	-.322
.280	-.121	.332	.088	-.355
.326	-.163	.324	.079	-.443
.371	-.345	.436	-.088	-.509
.392	-.420	.490	-1.008	-.630
.413	-.513	.544	-1.212	.177
.434	-.579	.571	-1.886	.644
.457	-.551	.580	-1.449	.723
.480	-.523	.580	-1.171	.686
.502	-.527	.570	-1.073	.649
.551	-.439	.560	-1.017	.635
.585	-.383	.544	-1.166	.686
.592	-.355	.526	-1.231	-1.046
.613	-.257	.431	-.892	-.700
.634	-.224	.319	-.585	-.322
.655	-.168	.184	-.353	-.331
.675	-.084	.058	-.190	-.233
.696	-.037	.076	-.074	-.075
.774	-.023	.211	.070	-.040
.852	-.028	.067	.005	-.005
.930	.061	.094	.033	.070

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-5.000	-3.334	-4.014	-4.153	-4.014
	.080	-1.873	-2.557	-3.986	-4.162	-2.782
	.130	-3.297	-3.046	-3.417	-3.707	-3.823
	.145	-10.192	-9.108	-8.747	-9.378	-10.269
	.155	-4.409	-4.583	-4.882	-5.031	-4.658
	.180	-2.972	-3.091	-3.575	-3.609	-3.818
	.220	-1.832	-2.247	-2.633	-2.810	-2.754
	.270	-1.378	-1.811	-2.287	-2.262	-2.282
	.400	-1.026	-1.393	-1.802	-1.965	-2.203
	.620	-.948	-1.591	-2.100	-2.402	-2.875
	.685	-2.944	-4.592	-5.424	-7.883	-14.054
	.693	-2.825	-4.992	-7.226	-8.682	-13.877
	.700	-1.690	-3.599	-5.555	-5.607	-10.638
	.720	-.765	-1.492	-2.446	-3.001	-4.994
	.750	-.527	-.795	-1.508	-2.011	-3.585
	.800	-.467	-.382	-.808	-1.175	-2.777
	.900	-.371	-.144	-.257	-.548	-1.942
	.980	-.279	.049	.079	-.381	-.901
Lower	.025	.733	.827	.817	.794	.616
	.120	.884	.822	.765	.720	.504
	.220	.847	.831	.822	.762	.550
	.300	.769	.768	.756	.701	.471
	.620	.801	.822	.775	.701	-.177
	.750	.865	.876	.765	.678	.509
	.850	.678	.746	.719	.641	.457
	.950	.458	.557	.537	.423	.266

TABLE 8 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = [illegible]°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 22.8°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.238	.839	-.446	.046
.053	-.345	.660	-.567	.132
.100	-.215	.419	-.599	.461
.145	-.173	.321	-.567	.416
.189	-.084	.339	-.516	-.351
.234	-.061	.382	-.316	.425
.280	-.135	.377	.135	-.453
.326	-.210	.405	.111	-.573
.371	-.443	.490	-.367	-.647
.392	-.520	.550	-1.041	.869
.413	-.622	.627	-1.310	.176
.434	-.221	.636	-2.100	.684
.457	-.579	.640	-1.640	.744
.480	-.519	.635	-1.347	.712
.502	-.495	.600	-1.198	.675
.551	-.355	.580	-1.092	.643
.585	-.294	.566	-1.236	.666
.592	-.271	.547	-1.570	-1.470
.613	-.191	.443	-1.157	-.900
.634	-.161	.311	-.674	-.453
.655	-.145	.189	-.367	-.324
.675	-.079	.047	-.177	-.129
.696	-.031	.090	-.074	[illegible]
.774	-.050	.226	.246	.139
.852	-.020	.104	-.019	-.029
.930	.047	.113	.060	.088

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-8.764	4.186	-4.378	-4.131	-3.949
	.080	-2.051	3.986	-4.475	-4.148	-2.586
	.140	-3.324	3.073	3.504	3.121	-2.605
	.145	-9.734	8.371	-6.050	-6.564	7.039
	.155	-4.135	4.487	-4.172	-3.715	-2.931
	.180	-2.766	3.134	-3.259	-2.750	-2.432
	.220	-1.717	2.277	-2.293	-2.057	-1.573
	.270	-1.456	1.810	-1.909	-1.649	-1.209
	.400	-1.035	1.372	1.442	-1.310	-1.144
	.670	-.957	1.504	1.322	-1.050	-1.069
	.685	-3.810	3.714	-.573	-2.053	-1.918
	.690	-3.832	4.068	2.102	-2.276	-1.578
	.700	-2.459	3.002	-1.812	-1.538	-1.396
	.720	-1.213	1.296	1.091	-1.092	-1.022
	.750	-.783	-.787	-1.026	-1.106	-1.055
	.800	-.531	-.485	-.952	-1.022	-1.041
	.900	-.375	-.292	-.878	-.948	-.994
	.980	-.270	-.028	-.740	-.910	-.924
Lower	.025	.797	.872	.823	.818	.653
	.120	.907	.863	.772	.743	.621
	.220	.875	.877	.818	.780	.658
	.300	.792	.806	.758	.748	.569
	.620	.788	.853	.758	.715	.247
	.750	.884	.896	.754	.743	.523
	.850	.696	.749	.619	.590	.481
	.950	.504	.528	.259	.241	.275

X

L-927

TABLE 9

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 60°$; $\delta_{a,L} = 60°$; $\delta_{a,R} = 60°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface	Surface	x/c	0.221	0.426	0.640	0.800	0.918
	Fuselage						Wing, flap, or aileron				
$\alpha = -1.9°$											
.032	.273	.290	.277	.296	Upper	.010	.876	.821	.773	.795	.781
.053	.044	.070	.066	.052		.080	.321	.220	.129	.119	.159
.100	-.097	-.044	-.092	-.082		.130	-.594	-.883	-1.022	-.983	-.988
.145	-.088	-.088	-.053	-.034		.145	-4.983	-4.926	-4.791	-5.361	-4.958
.189	-.009	-.013	.000	.017		.155	-1.953	-2.050	-2.112	-2.064	-1.553
.234	-.031	.061	-.018	.030		.180	-1.509	-1.352	-1.554	-1.431	-1.482
.280	-.057	.088	.004	.047		.220	-.944	-1.041	-1.146	-1.256	-1.045
.326	-.057	.110	-.035	.099		.270	-.765	-.957	-1.052	-.962	-.860
.371	-.124	.189	-.132	.163		.400	-.821	-1.019	-1.150	-1.089	-.975
.392	-.170	.200	-.031	.404		.620	-1.466	-1.835	-1.743	-1.493	-1.531
.413	-.194	.233	-.224	.266		.685	-8.077	-9.269	-7.280	-9.852	-8.499
.434	-.247	.259	-.492	.133		.693	-7.342	-9.176	-8.203	-9.778	-7.009
.457	-.313	.300	-.637	.142		.700	-4.491	-6.388	-5.872	-5.822	-5.011
.480	-.371	.305	-.571	.193		.720	-2.098	-2.700	-2.254	-2.586	-2.157
.502	-.459	.305	-.623	.258		.750	-1.517	-1.585	-1.339	-1.607	-1.645
.551	-.512	.295	-.852	.258		.800	-1.299	-.856	-.816	-.988	-1.367
.585	-.494	.290	-1.045	.185		.900	-.821	-.176	-.498	-.584	-1.147
.592	-.463	.215	-1.256	-1.112		.980	-.068	.329	-.219	-.123	-.909
.613	-.344	.097	-1.014	-1.515	Lower	.025	.150	.215	.309	.290	.062
.634	-.282	.004	-.711	-.519		.120	.077	.211	.279	.228	.018
.655	-.221	-.026	-.465	-.219		.220	.073	.176	.240	.206	.026
.675	-.141	-.075	-.303	-.103		.300	.175	.145	.176	.158	-.022
.696	-.066	-.044	-.711	-.064		.620	.286	.369	.262	.215	.075
.774	-.035	.031	-.075	.004		.750	.470	.505	.352	.316	.357
.852	-.022	-.031	.004	-.120		.850	.774	.667	.502	.413	.459
.930	.093	-.228	.105	-.279		.950	.718	.716	.567	.448	.353
$\alpha = 5.3°$											
.032	.082	.467	.132	.306	Upper	.010	.490	.450	.346	.321	.437
.053	-.132	.227	-.070	.075		.080	-.294	-.383	-.514	-.593	-.419
.100	-.201	.053	-.189	-.115		.130	-1.531	-1.807	-2.074	-2.042	-2.010
.145	-.150	-.004	-.158	-.084		.145	-7.127	-7.078	-6.926	-7.644	-7.310
.189	-.073	.045	-.110	-.031		.155	-2.956	-3.125	-3.292	-3.271	-2.689
.234	-.091	.116	-.048	-.044		.180	-2.172	-2.061	-2.344	-2.226	-2.315
.280	-.105	.147	.048	-.031		.220	-1.407	-1.527	-1.733	-1.840	-1.668
.326	-.123	.156	.035	-.035		.270	-1.077	-1.327	-1.507	-1.436	-1.353
.371	-.232	.240	-.171	.018		.400	-.953	-1.229	-1.493	-1.436	-1.381
.392	-.280	.290	-.312	.146		.620	-1.184	-1.865	-1.963	-1.875	-2.037
.413	-.337	.343	-.610	.261		.685	-3.410	-8.810	-7.679	-11.170	-12.751
.434	-.410	.387	-.935	.368		.693	-2.586	-8.716	-8.561	-11.341	-10.873
.457	-.447	.400	-.887	.474		.700	-1.442	-6.001	-6.119	-6.731	-7.929
.480	-.510	.450	-.786	.523		.720	-.743	-2.506	-2.397	-3.122	-3.600
.502	-.597	.415	-.760	.523		.750	-.712	-1.407	-1.467	-1.936	-2.579
.551	-.583	.470	-.856	.563		.800	-.650	-.752	-1.001	-1.203	-2.233
.585	-.547	.467	-.900	.620		.900	-.583	-.249	-.744	-.760	-1.973
.592	-.515	.405	-.869	-.868		.980	-.534	.165	-.492	-.193	-1.522
.613	-.387	.240	-.707	-.545	Lower	.025	.200	.463	.359	.566	.164
.634	-.342	.089	-.558	-.634		.120	.476	.507	.616	.566	.424
.655	-.287	-.022	-.421	-.514		.220	.690	.654	.727	.641	.606
.675	-.173	-.125	-.285	-.244		.300	.659	.672	.669	.623	.520
.696	-.114	-.089	-.176	-.093		.620	.783	.783	.767	.724	.027
.774	-.036	.040	.000	.000		.750	.877	.846	.793	.751	.611
.852	-.041	.031	-.004	-.155		.850	.712	.748	.700	.676	.533
.930	.055	-.174	.079	-.204		.950	.481	.601	.439	.487	.182
$\alpha = 9.1°$											
.032		.569	.031	.297	Upper	.010	.180	-.041	-.737	-1.029	-.216
.053	-.171	.352	-.170	.058		.080	-.683	-.790	-.988	-1.109	-.890
.100	-.189	.149	-.286	-.139		.130	-2.094	-2.375	-2.732	-2.670	-2.723
.145	-.153	.068	-.242	-.126		.145	-8.312	-8.230	-8.043	-8.810	-8.429
.189	-.072	.104	-.197	-.081		.155	-3.505	-3.724	-3.963	-3.962	-3.302
.234	-.117	.176	-.081	-.099		.180	-2.426	-2.438	-2.763	-2.670	-2.714
.280	-.117	.199	.076	-.117		.220	-1.550	-1.792	-2.040	-2.138	-1.914
.326	-.148	.203	.018	-.126		.270	-1.191	-1.530	-1.734	-1.677	-1.550
.371	-.261	.293	-.242	-.112		.400	-1.038	-1.336	-1.622	-1.606	-1.555
.392	-.312	.340	-.492	-.040		.620	-1.168	-1.901	-2.044	-2.129	-2.188
.413	-.377	.402	-.827	.216		.685	-2.768	-8.582	-7.540	-11.489	-12.608
.434	-.436	.460	-1.127	.526		.693	-2.031	-8.428	-8.420	-11.565	-10.689
.457	-.458	.500	-1.011	.620		.700	-1.213	-5.774	-6.012	-6.941	-7.764
.480	-.490	.505	-.894	.589		.720	-.710	-2.379	-2.359	-3.229	-3.446
.502	-.584	.505	-.859	.562		.750	-.683	-1.314	-1.469	-2.035	-2.503
.551	-.539	.500	-.930	.638		.800	-.634	-.713	-1.060	-1.301	-2.269
.585	-.503	.492	-.921	.669		.900	-.557	-.271	-.800	-.859	-2.058
.592	-.467	.438	-.881	-.849		.980	-.539	.126	-.571	-.304	-1.712
.613	-.359	.298	-.733	-.499	Lower	.025	.391	.637	.643	.675	.526
.634	-.337	.099	-.635	-.611		.120	.759	.813	.755	.725	.562
.655	-.288	-.032	-.470	-.530		.220	.777	.790	.750	.733	.607
.675	-.171	-.113	-.309	-.279		.300	.683	.722	.710	.662	.503
.696	-.130	-.032	-.197	-.121		.620	.809	.862	.822	.716	.045
.774	-.063	.126	-.009	.045		.750	.930	.939	.809	.742	.566
.852	-.054	.068	-.013	-.090		.850	.714	.840	.741	.671	.481
.930	.040	-.050	.049	-.090		.950	.467	.632	.454	.492	.103

TABLE 9 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 60°; $\delta_{a,L}$ = 60°; $\delta_{a,R}$ = 60°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 12.9°											
.032	-.062	.637	-.089	.283	Upper	.010	-.365	-1.761	-2.130	-2.352	-1.972
.053	-.229	.397	-.268	.076		.080	-1.149	-1.106	-1.240	-1.391	-1.200
.100	-.194	.208	-.362	-.189		.130	-2.640	-[illegible].907	-3.190	-3.122	-3.211
.145	-.150	.122	-.322	-.180		.145	-9.010	-[illegible].078	-8.717	-9.526	-9.082
.189	-.088	.153	-.277	-.126		.155	-3.891	-[illegible].248	-4.439	-4.499	-3.731
.234	-.115	.212	-.098	-.166		.180	-2.653	-[illegible].785	-3.105	-3.041	-3.008
.280	-.115	.239	.103	-.175		.220	-1.696	-[illegible].013	-2.287	-2.361	-2.108
.326	-.150	.244	.045	-.225		.270	-1.291	- .666	-1.901	-1.874	-1.729
.371	-.273	.348	-.250	-.252		.400	-1.068	- .377	-1.662	-1.704	-1.689
.392	-.325	.398	-.662	-.238		.620	-1.104	- .801	-2.049	-2.174	-2.170
.413	-.397	.447	-1.002	.234		.685	-2.600	-[illegible].173	-7.081	-10.930	-11.852
.434	-.450	.492	-1.342	.589		.693	-1.963	-[illegible].006	-7.948	-10.939	-10.034
.457	-.463	.505	-1.154	.678		.700	-1.144	-[illegible].749	-5.648	-6.570	-7.278
.480	-.507	.510	-.997	.643		.720	-.605	-[illegible].882	-2.224	-3.081	-3.255
.502	-.556	.518	-.930	.620		.750	-.588	[illegible].962	-1.393	-1.914	-2.421
.551	-.512	.515	-.966	.674		.800	-.543	[illegible].465	-1.011	-1.257	-2.139
.585	-.463	.515	-.970	.692		.900	-.467	-.330	-.836	-.868	-1.932
.592	-.432	.447	-.988	-.962		.980	-.427	-.144	-.607	-.389	-1.628
.613	-.331	.311	-.778	-.373	Lower	.025	.534	.754	.777	.760	.591
.634	-.318	.122	-.572	-.481		.120	.819	.835	.773	.716	.525
.655	-.256	-.027	-.389	-.485		.220	.797	.808	.804	.756	.560
.675	-.168	-.126	-.264	-.288		.300	.694	.758	.737	.689	.498
.696	-.115	-.036	-.174	-.117		.620	.819	.871	.822	.720	-.035
.774	-.057	.158	.004	.072		.750	.904	.943	.800	.733	.538
.852	-.062	.086	-.045	-.022		.850	.703	.808	.746	.684	.441
.930	.009	.023	.009	[illegible]		.950	.476	.596	.467	.479	.146
α = 16.8°											
.032	-.125	.717	-.230	.195	Upper	.010	-3.901	- .853	-3.415	-3.643	-3.394
.053	-.259	.503	-.370	.005		.080	-1.687	- .760	-3.012	-3.462	-2.111
.100	-.197	.281	-.460	-.218		.130	-3.130	- .039	-3.288	-3.363	-3.542
.145	-.148	.204	-.429	-.245		.145	-9.901	-[illegible].216	-8.790	-9.281	-9.763
.189	-.089	.222	-.370	-.195		.155	-4.282	-[illegible].526	-4.758	-4.826	-4.258
.234	-.094	.263	-.144	-.259		.180	-2.866	- .016	-3.420	-3.417	-3.466
.280	-.121	.290	.122	-.272		.220	-1.810	-[illegible].200	-2.499	-2.659	-2.531
.326	-.152	.308	.072	-.363		.270	-1.356	-[illegible].787	-2.091	-2.149	-2.097
.371	-.295	.408	-.284	-.399		.400	-1.052	- .415	-1.769	-1.855	-2.048
.392	-.375	.460	-.840	-.476		.620	-.989	- .773	-2.114	-2.456	-2.442
.413	-.456	.531	-1.106	.204		.685	-2.744	-[illegible].245	-6.826	-9.918	-12.867
.434	-.519	.558	-1.594	.649		.693	-2.236	-[illegible].100	-7.738	-9.945	-10.921
.457	-.496	.560	-1.381	.726		.700	-1.265	-[illegible].100	-5.542	-6.027	-7.952
.480	-.496	.560	-1.115	.698		.720	-.612	-[illegible].583	-2.222	-2.961	-3.618
.502	-.537	.560	-1.043	.662		.750	-.522	-.776	-1.365	-1.905	-2.701
.551	-.465	.560	-1.097	.685		.800	-.517	-.395	-.916	-1.205	-2.330
.585	-.425	.526	-1.201	.726		.900	-.417	-.345	-.803	-.799	-2.017
.592	-.394	.481	-1.124	-1.093		.980	-.386	-.181	-.544	-.537	-1.601
.613	-.291	.336	-.790	-.363	Lower	.025	.680	.835	.825	.804	.599
.634	-.277	.145	-.555	-.440		.120	.889	.844	.780	.709	.501
.655	-.233	-.009	-.375	-.440		.220	.835	.835	.816	.763	.568
.675	-.130	-.122	-.235	-.259		.300	.762	.789	.767	.718	.474
.696	-.081	-.036	-.131	-.095		.620	.839	.884	.835	.727	-.134
.774	-.045	.050	.041	.100		.750	.921	.907	.803	.736	.519
.852	-.049	.100	-.041	.018		.850	.757	.825	.771	.700	.429
.930	.009	.100	[illegible]	.068		.950	.535	.608	.526	.465	.170

TABLE 10

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = -1.9°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.253	.303	.263	.297
.053	.039	.059	.051	.058
.100	-.130	-.053	-.115	-.103
.145	-.097	-.112	-.071	-.045
.189	-.045	-.026	-.013	-.006
.234	-.071	.046	.006	.006
.280	-.071	.072	[illegible]	.039
.326	-.078	.099	-.019	.065
.371	-.117	.151	-.122	.168
.392	-.163	.175	-.026	.374
.413	-.208	.198	-.224	.239
.434	-.260	.198	-.468	.110
.457	-.292	.207	-.603	.084
.480	-.364	.217	-.545	.136
.502	-.455	.226	-.590	.168
.551	-.487	.244	-.750	.232
.585	-.461	.257	-.878	.168
.592	-.448	.211	-1.006	-.729
.613	-.299	.079	-.795	-.798
.634	-.279	.053	-.558	-.858
.655	-.234	.026	-.378	-.329
.675	-.130	-.033	-.250	-.136
.696	-.084	-.013	-.173	-.071
.774	-.071	-.007	-.058	.006
.852	-.032	[illegible]	-.026	-.136
.930	.078	-.178	.071	-.232

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.859	.797	.794	.769	.793
	.080	.321	.165	.136	.051	.032
	.130	-.583	-.909	-1.052	-1.115	-1.332
	.145	-4.942	-5.018	-4.898	-5.686	-5.964
	.155	-1.897	-2.107	-2.168	-2.314	-2.287
	.180	-1.429	-1.390	-1.594	-1.609	-2.001
	.220	-.897	-1.080	-1.226	-1.385	-1.507
	.270	-.737	-.962	-1.091	-1.135	-1.273
	.400	-.801	-1.001	-1.187	-1.250	-1.286
	.620	-1.308	-1.693	-1.813	-1.840	-1.884
	.685	-6.538	-7.126	-7.086	-9.808	-10.980
	.693	-6.429	-7.857	-8.686	-10.096	-9.823
	.700	-4.051	-5.664	-6.608	-6.756	-7.335
	.720	-1.917	-2.509	-2.852	-3.090	-3.060
	.750	-1.218	-1.515	-1.723	-2.000	-1.943
	.800	-.936	-.876	-.962	-1.250	-1.397
	.900	-.673	-.244	-.297	-.718	-1.286
	.980	-.147	.237	.200	-.160	-.812
Lower	.025	.115	.165	.200	.224	.227
	.120	.045	.158	.194	.205	.227
	.220	.026	.132	.161	.192	.221
	.300	.115	.125	.142	.186	.208
	.620	.269	.244	.252	.276	.221
	.750	.333	.349	.387	.359	.279
	.850	.551	.501	.561	.513	.390
	.950	.538	.580	.549	.481	.338

$\alpha = 5.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.066	.433	.123	.299
.053	-.126	.204	-.078	.058
.100	-.225	.045	-.214	-.130
.145	-.166	-.025	-.182	-.097
.189	-.093	.025	-.104	-.045
.234	-.126	.089	.019	-.052
.280	-.133	.108	.065	-.065
.326	-.139	.127	.013	-.052
.371	-.245	.217	-.214	-.013
.392	-.298	.258	-.305	.097
.413	-.351	.299	-.637	.253
.434	-.444	.331	-.975	.299
.457	-.464	.343	-.910	.429
.480	-.511	.355	-.806	.468
.502	-.603	.367	-.780	.455
.551	-.577	.391	-.890	.468
.585	-.550	.408	-.923	.526
.592	-.504	.388	-.871	-.851
.613	-.338	.287	-.702	-.615
.634	-.338	.178	-.559	-.585
.655	-.298	.070	-.422	-.546
.675	-.186	-.013	-.299	-.260
.696	-.139	-.019	-.167	-.091
.774	-.060	.083	[illegible]	-.127
.852	-.066	.045	-.026	-.162
.930	.033	-.115	.045	-.136

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.465	.357	.253	.188	.139
	.080	-.344	-.446	-.565	-.741	-.776
	.130	-1.541	-1.859	-2.183	-2.358	-2.712
	.145	-7.138	-6.966	-7.147	-8.407	-8.919
	.155	-2.961	-3.133	-3.424	-3.768	-3.747
	.180	-2.191	-2.089	-2.449	-2.592	-3.084
	.220	-1.433	-1.573	-1.819	-2.092	-2.314
	.270	-1.089	-1.344	-1.585	-1.702	-1.936
	.400	-1.000	-1.235	-1.527	-1.696	-1.830
	.620	-1.229	-1.828	-2.118	-2.417	-2.467
	.685	-3.623	-6.915	-7.692	-11.168	-13.190
	.693	-3.216	-7.501	-9.343	-11.681	-11.916
	.700	-1.923	-5.362	-7.179	-7.842	-9.005
	.720	-.866	-2.394	-3.131	-3.794	-3.946
	.750	-.726	-1.452	-1.936	-2.508	-2.606
	.800	-.694	-.841	-1.124	-1.624	-1.910
	.900	-.567	-.306	-.429	-.871	-1.678
	.980	-.509	.013	.091	-.260	-1.147
Lower	.025	.204	.363	.455	.604	.603
	.120	.344	.478	.546	.682	.716
	.220	.567	.630	.741	.728	.756
	.300	.605	.624	.650	.682	.690
	.620	.662	.713	.754	.747	.723
	.750	.745	.783	.819	.793	.769
	.850	.541	.656	.689	.637	.497
	.950	.331	.516	.533	.409	.126

$\alpha = 12.8°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.084	.602	-.104	.259
.053	-.258	.353	-.260	.046
.100	-.207	.183	-.351	-.199
.145	-.161	.092	-.318	-.192
.189	-.110	.118	-.286	-.153
.234	-.136	.183	-.052	-.179
.280	-.155	.203	.084	-.206
.326	-.181	.190	.013	-.252
.371	-.310	.207	-.286	-.259
.392	-.365	.370	-.617	-.259
.413	-.419	.432	-.962	.186
.434	-.478	.464	-1.325	.550
.457	-.503	.468	-1.150	.630
.480	-.523	.473	-1.007	.603
.502	-.587	.477	-.942	.564
.551	-.510	.485	-.988	.603
.585	-.478	.491	-.975	.650
.592	-.465	.432	-.962	-.948
.613	-.310	.327	-.715	-.600
.634	-.323	.222	-.520	-.431
.655	-.271	.098	-.370	-.477
.675	-.187	-.033	-.221	-.338
.696	-.123	-.007	-.123	-.159
.774	-.095	.033	-.077	-.102
.852	-.077	.072	-.032	-.046
.930	.006	.013	.005	.027

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.340	-2.060	-2.394	-2.969	-3.246
	.080	-1.119	-1.204	-1.353	-1.676	-2.091
	.130	-2.584	-2.963	-3.389	-3.417	-3.898
	.145	-9.020	-9.197	-9.125	-10.194	-10.900
	.155	-3.859	-4.311	-4.755	-5.003	-5.156
	.180	-2.636	-2.832	-3.336	-3.430	-4.040
	.220	-1.720	-2.080	-2.467	-2.703	-3.020
	.270	-1.328	-1.707	-2.076	-2.170	-2.568
	.400	-1.132	-1.400	-1.817	-1.949	-2.284
	.620	-1.282	-1.962	-2.334	-2.677	-2.672
	.685	-2.872	-6.430	-7.540	-10.954	-12.558
	.693	-2.250	-6.548	-8.999	-11.370	-11.242
	.700	-1.380	-4.664	-6.916	-7.621	-8.447
	.720	-.661	-1.982	-3.064	-3.716	-3.756
	.750	-.569	-1.145	-1.916	-2.482	-2.485
	.800	-.504	-.608	-1.121	-1.650	-1.968
	.900	-.432	-.249	-.378	-.936	-1.897
	.980	-.556	-.301	-.046	-.539	-1.446
Lower	.025	.497	.726	.736	.786	.749
	.120	.818	.805	.763	.773	.800
	.220	.785	.778	.802	.793	.781
	.300	.680	.700	.710	.728	.761
	.620	.752	.791	.802	.760	.723
	.750	.831	.883	.816	.786	.761
	.850	.602	.713	.683	.643	.510
	.950	.373	.517	.524	.390	.103

TABLE 10 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					α = 18.5°						
.032	-.184	.713	-.305	.159	Upper	.010	-5.914	-3.570	-4.198	-4.678	-5.038
.053	-.323	.498	-.442	-.033		.080	-1.877	-2.508	-4.218	-4.671	-5.157
.100	-.231	.303	-.520	-.272		.130	-3.293	-3.154	-3.269	-3.495	-4.189
.145	-.178	.208	-.481	-.325		.145	-10.169	-9.494	-8.607	-9.122	-8.739
.189	-.086	.235	-.442	-.279		.155	-4.386	-4.788	-4.847	-5.055	-4.854
.234	-.099	.282	-.058	-.338		.180	-2.931	-3.228	-3.574	-3.671	-4.044
.280	-.138	.296	.130	-.378		.220	-1.831	-2.367	-2.646	-2.839	-3.115
.326	-.191	.336	.058	-.458		.270	-1.383	-1.903	-2.175	-2.326	-2.852
.371	-.375	.430	-.390	-.511		.400	-1.093	-1.479	-1.771	-1.962	-2.727
.392	-.451	.491	-.845	-.650		.620	-1.067	-1.775	-2.076	-2.079	-2.180
.413	-.527	.551	-1.150	.172		.685	-2.852	-4.781	-5.703	-6.932	-5.032
.434	-.580	.585	-1.787	.637		.693	-2.687	-5.251	-7.195	-7.387	-4.235
.457	-.573	.577	-1.390	.723		.700	-1.666	3.712	-5.524	-4.775	-3.649
.480	-.553	.568	-1.156	.683		.720	-.777	1.587	-2.460	-2.547	-2.325
.502	-.586	.560	-1.065	.643		.750	-.586	-.881	-1.538	-1.845	-2.101
.551	-.487	.543	-1.046	.597		.800	-.533	-.464	-.882	-1.390	-1.936
.585	-.435	.531	-1.156	.670		.900	-.382	-.249	-.318	-1.040	-1.798
.592	-.428	.491	-1.130	-1.121		.980	-.435	-.256	-.020	-.773	-1.607
.613	-.257	.390	-.825	-.500	Lower	.025	.718	.861	.829	.799	.738
.634	-.263	.282	-.559	-.351		.120	.883	.861	.796	.793	.810
.655	-.224	.148	-.325	-.338		.220	.843	.847	.836	.825	.836
.675	-.119	[illegible]	-.182	-.239		.300	.757	.787	.756	.780	.790
.696	-.072	.040	-.078	-.099		.620	.771	.861	.789	.767	.757
.774	-.033	.074	-.055	-.043		.750	.850	.894	.829	.767	.771
.852	-.059	.108	-.032	.013		.850	.659	.760	.683	.604	.527
.930	-.007	.114	-.019	.133		.950	.448	.524	.504	.286	.020
					α = 23.0°						
.032	-.227	.741	-.448	.007	Upper	.010	-8.420	-3.788	-4.103	-3.767	-3.586
.053	-.331	.520	-.560	-.178		.080	-2.030	-3.424	-4.228	-3.833	-3.404
.100	-.188	.383	-.632	-.395		.130	-3.312	-2.748	-3.260	-3.141	-2.228
.145	-.175	.279	-.566	-.441		.145	-9.796	-7.887	-6.408	-4.841	-4.840
.189	-.091	.299	-.514	-.375		.155	-4.147	-4.119	-3.767	-2.865	-2.131
.234	-.045	.318	-.092	-.415		.180	-2.718	-2.787	-2.779	-2.154	-1.832
.280	-.110	.357	.119	-.461		.220	-1.676	-2.001	-1.963	-1.607	-1.306
.326	-.195	.357	.040	-.573		.270	-1.369	-1.579	-1.574	-1.344	-1.143
.371	-.409	.474	-.448	-.659		.400	-1.035	-1.124	-1.245	-1.172	-1.143
.392	-.491	.536	-.922	-.850		.620	-.988	-1.052	-1.067	-1.060	-1.033
.413	-.572	.598	-1.258	.145		.685	-4.060	-1.858	-.283	-1.469	-1.923
.434	-.585	.604	-2.002	.626		.693	-4.120	-2.079	-1.594	-1.607	-1.494
.457	-.526	.592	-1.554	.698		.700	-2.698	-1.618	-1.409	-1.218	-1.325
.480	-.461	.580	-1.311	.659		.720	-1.409	-.871	-.948	-.942	-.955
.502	-.455	.568	-1.126	.632		.750	-.942	-.793	-.915	-.948	-.962
.551	-.212	.544	-1.060	.599		.800	-.668	-.793	-.856	-.883	-.897
.585	-.247	.526	-1.153	.632		.900	-.454	-.747	-.784	-.738	-.767
.592	-.214	.487	-1.567	-1.508		.980	-.374	-.663	-.731	-.711	-.721
.613	-.097	.377	-1.284	-.600	Lower	.025	.821	.845	.836	.856	.754
.634	-.130	.266	-.757	-.560		.120	.942	.838	.803	.836	.832
.655	-.136	.130	-.441	-.310		.220	.888	.845	.823	.850	.845
.675	-.058	-.005	-.250	-.112		.300	.808	.767	.764	.803	.812
.696	-.032	.039	-.151	-.013		.620	.808	.819	.777	.803	.786
.774	-.039	.081	[illegible]	-.003		.750	.901	.858	.836	.803	.793
.852	-.104	.123	-.066	.007		.850	.688	.669	.599	.632	.552
.930	-.006	.130	.007	.105		.950	.487	.344	.217	.250	.162

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TABLE 11

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 0.0$; $h_d/c = 0.0$

$C_{\mu,k} = 0.019$ $C_{\mu,f} = 0.019$ $C_{\mu,a} = 0.007$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)	Surface	x/c	0.221 (Wing, flap, or aileron)	0.426	0.640	0.800	0.918
$\alpha = -2.1°$											
.032	.273	.327	.293	.288	Upper	.010	.873	.804	.782	.774	.775
.053	.025	.082	.075	.051		.080	.291	.162	.103	.037	.050
.100	-.112	-.031	-.094	-.115		.130	-.588	-.886	-1.103	-1.105	-1.321
.145	-.105	-.069	-.081	-.071		.145	-5.352	-4.996	-5.218	-6.125	-6.278
.189	-.037	[illegible]	-.006	.006		.155	-1.898	-1.917	-2.064	-2.079	-1.780
.234	-.062	.057	.025	.026		.180	-1.594	-1.282	-1.609	-1.548	-2.066
.280	-.074	.082	.006	.032		.220	-.905	-1.006	-1.199	-1.405	-1.476
.326	-.068	.101	.012	.083		.270	-.746	-.949	-1.141	-1.080	-1.234
.371	-.149	.157	-.031	.154		.400	-.816	-.999	-1.263	-1.255	-1.303
.392	-.177	.189	-.031	.385		.620	-1.354	-1.678	-1.846	-1.973	-1.861
.413	-.205	.220	-.237	.250		.685	-7.300	-7.190	-6.795	-9.990	-11.507
.434	-.254	.239	-.493	.135		.693	-7.287	-6.088	-9.558	-11.064	-10.527
.457	-.310	.248	-.599	.122		.700	-4.491	-5.926	-7.295	-6.987	-7.773
.480	-.372	.257	-.549	.167		.720	-2.056	-2.501	-2.910	-3.147	-3.046
.502	-.459	.265	-.599	.205		.750	-1.297	-1.552	-1.801	-2.073	-1.929
.551	-.484	.282	-.787	.256		.800	-.873	-.962	-1.013	-1.761	-1.359
.585	-.478	.295	-.930	.244		.900	-.727	-.333	-.378	-.606	-1.241
.592	-.440	.258	-1.086	-.981		.980	-.171	.170	.141	-.006	-.707
.613	-.366	.182	-.855	-.910	Lower	.025	.152	.182	.212	.231	.248
.634	-.292	.132	-.593	-.897		.120	.076	.182	.205	.219	.230
.655	-.230	.063	-.406	-.340		.220	.038	.151	.179	.206	.236
.675	-.130	.025	-.281	-.141		.300	.152	.163	.147	.187	.230
.696	-.074	.031	-.175	-.090		.620	.272	.289	.276	.287	.248
.774	-.050	.057	-.056	-.125		.750	.392	.408	.423	.412	.304
.852	-.043	.013	-.025	-.160		.850	.607	.522	.526	.574	.422
.930	.068	-.182	.094	-.276		.950	.557	.559	.577	.562	.366
$\alpha = 5.0°$											
.032	.072	.455	.123	.331	Upper	.010	.449	.323	.159	.052	-.052
.053	-.118	.209	-.078	.096		.080	-.365	-.481	-.675	-.871	-.903
.100	-.177	.070	-.214	-.096		.130	-1.622	-1.929	-2.286	-2.475	-2.970
.145	-.157	.013	-.188	-.076		.145	-7.737	-7.452	-7.654	-9.245	-9.792
.189	-.085	.063	-.136	-.045		.155	-3.058	-3.139	-3.400	-3.690	-3.513
.234	-.118	.108	-.019	-.038		.180	-2.385	-2.107	-2.522	-2.618	-3.362
.280	-.124	.127	.058	-.045		.220	-1.455	-1.575	-1.872	-2.241	-2.420
.326	-.157	.152	.006	-.051		.270	-1.154	-1.392	-1.649	-1.722	-2.041
.371	-.275	.234	-.188	-.032		.400	-1.083	-1.316	-1.668	-1.761	-1.936
.392	-.330	.282	-.364	.096		.620	-1.571	-1.967	-2.197	-2.527	-2.551
.413	-.386	.329	-.687	.191		.685	-7.141	-7.806	-7.622	-11.889	-14.776
.434	-.445	.380	-.994	.376		.693	-6.885	-8.837	-10.386	-13.085	-13.370
.457	-.491	.393	-.949	.497		.700	-4.218	-6.478	-7.972	-8.387	-9.956
.480	-.549	.406	-.864	.507		.720	-2.013	-2.733	-3.286	-3.957	-4.173
.502	-.621	.418	-.864	.478		.750	-1.513	-1.708	-2.082	-2.657	-2.760
.551	-.621	.443	-1.078	.522		.800	-1.090	-1.063	-1.216	-1.722	-2.041
.585	-.595	.462	-1.182	.586		.900	-.590	-.386	-.522	-.903	-1.799
.592	-.563	.373	-1.299	-1.184		.980	-.224	.070	.025	-.253	-1.125
.613	-.412	.266	-1.059	-.771	Lower	.025	.167	.474	.548	.617	.641
.634	-.373	.247	-.786	-.707		.120	.487	.588	.656	.734	.798
.655	-.294	.152	-.520	-.159		.220	.718	.683	.745	.760	.772
.675	-.190	.019	-.357	-.064		.300	.622	.633	.675	.689	.720
.696	-.137	.025	-.221	-.064		.620	.705	.765	.751	.741	.733
.774	-.078	.076	-.065	-.006		.750	.801	.797	.834	.806	.785
.852	-.046	.044	-.019	-.127		.850	.609	.696	.656	.643	.530
.930	.052	-.108	.084	-.210		.950	.474	.569	.535	.461	.164
$\alpha = 12.5°$											
.032	-.086	.621	-.113	.239	Upper	.010	-.464	-2.191	-2.633	-3.289	-3.707
.053	-.245	.386	-.279	.013		.080	-1.302	-1.210	-1.525	-2.062	-2.878
.100	-.212	.233	-.378	-.192		.130	-2.872	-3.055	-3.462	-3.614	-4.058
.145	-.159	.118	-.345	-.212		.145	-9.969	-9.589	-9.675	-11.267	-12.009
.189	-.099	.150	-.298	-.159		.155	-4.134	-4.291	-4.688	-5.086	-5.020
.234	-.119	.229	-.266	-.199		.180	-2.976	-2.839	-3.445	-3.614	-4.489
.280	-.153	.229	.073	-.212		.220	-1.825	-2.100	-2.520	-2.938	-3.289
.326	-.192	.242	.007	-.272		.270	-1.393	-1.766	-2.149	-2.308	-2.792
.371	-.332	.360	-.279	-.305		.400	-1.171	-1.478	-1.923	-2.129	-2.546
.392	-.405	.412	-.703	-.312		.620	-1.518	-2.119	-2.507	-2.858	-2.977
.413	-.477	.464	-1.094	.286		.685	-4.284	-7.457	-7.759	-12.719	.000
.434	-.517	.510	-1.485	.557		.693	-3.303	-7.679	-10.338	-13.362	-13.893
.457	-.550	.511	-1.313	.659		.700	-1.949	-5.606	-7.977	-8.727	-10.351
.480	-.577	.512	-1.114	.610		.720	-.896	-2.283	-3.322	-4.211	-4.423
.502	-.650	.513	-1.054	.557		.750	-.720	-1.367	-2.095	-2.838	-2.924
.551	-.590	.515	-1.074	.584		.800	-.661	-.791	-1.214	-1.844	-2.248
.585	-.564	.517	-1.107	.656		.900	-.517	-.262	-.484	-.988	-2.029
.592	-.537	.491	-1.147	-1.094		.980	-.621	-.196	-.126	-.617	-1.326
.613	-.405	.373	-.902	-.575	Lower	.025	.536	.739	.769	.816	.763
.634	-.358	.268	-.617	-.427		.120	.837	.811	.756	.789	.802
.655	-.312	.144	-.405	-.491		.220	.798	.805	.782	.822	.842
.675	-.172	.046	-.232	-.252		.300	.693	.726	.723	.756	.782
.696	-.139	.045	-.139	-.246		.620	.752	.798	.769	.776	.756
.774	-.027	.066	-.073	.060		.750	.837	.903	.842	.802	.782
.852	-.060	.085	-.007	-.082		.850	.628	.752	.710	.670	.544
.930	.020	.046	.020	.207		.950	.406	.621	.577	.471	.139

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TABLE 11 Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.019$ $C_{\mu,f} = 0.019$ $C_{\mu,a} = 0.007$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = 18.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.178	.741	-.303	.149
.053	-.310	.507	-.441	-.034
.100	-.191	.314	-.507	-.271
.145	-.178	.234	-.487	-.325
.189	-.099	.234	-.435	-.278
.234	-.112	.267	-.099	-.332
.280	-.145	.314	.119	-.359
.326	-.204	.300	.020	-.467
.371	-.389	.434	-.402	-.542
.392	-.464	.494	-.935	-.718
.413	-.540	.554	-1.258	.176
.434	-.606	.594	-1.989	.650
.457	-.573	.589	-1.508	.718
.480	-.586	.584	-1.271	.697
.502	-.626	.579	-1.192	.677
.551	-.520	.569	-1.225	.630
.585	-.481	.561	-1.396	.691
.592	-.435	.521	-1.383	-1.375
.613	-.329	.434	-.981	-.600
.634	-.296	.321	-.606	-.420
.655	-.270	.194	-.349	-.318
.675	-.138	.067	-.165	-.135
.696	-.092	.100	-.092	.007
.774	-.007	.114	-.059	.007
.852	-.072	.127	-.026	.007
.930	.007	.114	-.013	.122

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-6.771	-3.846	-4.740	-5.242	-5.638
	.080	-2.010	-2.938	-4.902	-5.335	-5.829
	.130	-3.492	-3.178	-3.548	-3.866	-4.742
	.145	-11.098	-9.816	-9.182	-10.155	-10.096
	.155	-4.614	-4.787	-5.024	-5.288	-5.104
	.180	-3.259	-3.265	-3.853	-3.991	-4.702
	.220	-2.003	-2.404	-2.871	-3.188	-3.655
	.270	-1.469	-1.990	-2.397	-2.555	-3.300
	.400	-1.175	-1.583	-2.025	-2.233	-2.990
	.620	-1.362	-2.077	-2.472	-2.786	-2.839
	.685	-4.046	-5.803	-6.413	-10.393	-11.301
	.693	-3.753	-6.317	-9.121	-11.459	-9.885
	.700	-2.270	-4.661	-7.069	-7.383	-7.435
	.720	-1.102	-1.923	-3.007	-3.728	-3.471
	.750	-.781	-1.102	-1.923	-2.621	-2.628
	.800	-.581	-.561	-1.083	-1.726	-2.325
	.900	-.401	-.147	-.366	-.863	-2.180
	.980	-.501	-.174	.014	-.566	-1.535
Lower	.025	.761	.841	.826	.784	.705
	.120	.921	.841	.779	.797	.823
	.220	.881	.861	.833	.830	.836
	.300	.801	.788	.792	.784	.817
	.620	.815	.841	.826	.790	.757
	.750	.895	.895	.860	.803	.797
	.850	.701	.775	.745	.665	.547
	.950	.501	.608	.596	.474	.112

$\alpha = 22.6°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.274	.802	-.464	.039
.053	-.394	.603	-.544	-.117
.100	-.207	.398	-.610	-.364
.145	-.194	.279	-.570	-.403
.189	-.093	.305	-.524	-.338
.234	-.073	.358	-.099	-.416
.280	-.140	.358	.159	-.455
.326	-.247	.391	.007	-.585
.371	-.461	.504	-.444	-.663
.392	-.555	.557	-1.021	-.923
.413	-.648	.610	-1.379	.195
.434	-.681	.630	-2.188	.689
.457	-.601	.620	-1.704	.767
.480	-.528	.611	-1.446	.715
.502	-.548	.601	-1.227	.702
.551	-.394	.582	-1.134	.676
.585	-.334	.570	-1.253	.682
.592	-.314	.531	-1.651	-1.553
.613	-.207	.418	-1.233	-.600
.634	-.187	.318	-.690	-.429
.655	-.167	.206	-.358	-.247
.675	-.080	.066	-.179	-.032
.696	-.040	.080	-.046	.065
.774	-.013	.093	.066	.065
.852	-.100	.106	.007	.065
.930	-.013	.146	.027	.162

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-9.905	-4.416	-4.873	-4.715	-4.848
	.080	-2.167	-4.092	-5.081	-4.781	-4.955
	.130	-3.471	-2.977	-3.599	-4.145	-3.799
	.145	-10.432	-8.873	-7.627	-6.200	-6.417
	.155	-4.254	-4.456	-4.333	-3.508	-2.898
	.180	-3.010	-3.117	-3.359	-2.772	-2.898
	.220	-1.897	-2.281	-2.449	-2.115	-2.050
	.270	-1.423	-1.870	-1.995	-1.718	-1.696
	.400	-1.106	-1.406	-1.540	-1.393	-1.583
	.620	-1.106	-1.565	-1.436	-1.273	-1.335
	.685	-4.287	-4.025	-1.299	-1.877	-2.678
	.693	-4.340	-4.682	-3.547	-2.168	-2.123
	.700	-2.720	-3.515	-2.729	-1.538	-1.823
	.720	-1.344	-1.472	-1.247	-1.094	-1.169
	.750	-.876	-.902	-1.104	-1.101	-1.162
	.800	-.560	-.537	-1.040	-1.021	-1.115
	.900	-.290	-.298	-1.020	-.922	-.995
	.980	-.356	-.192	-.936	-.836	-.935
Lower	.025	.803	.822	.845	.836	.728
	.120	.902	.862	.819	.849	.848
	.220	.869	.849	.845	.869	.875
	.300	.797	.796	.799	.829	.821
	.620	.784	.829	.793	.802	.775
	.750	.876	.882	.838	.802	.808
	.850	.718	.743	.650	.623	.568
	.950	.527	.544	.305	.232	.140

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TABLE 12 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = -14°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = -1.3°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.265	.312	.293	.299
.053	.045	.094	.062	.076
.100	-.097	-.037	-.081	-.076
.145	-.084	-.075	-.069	-.038
.189	.000	-.006	-.006	.000
.234	-.039	.056	-.006	.038
.280	-.039	.087	-.019	.045
.326	-.019	.081	-.037	.057
.371	-.103	.119	-.081	.140
.392	-.129	.113	.006	.325
.413	-.155	.106	-.131	-.376
.434	-.219	.144	-.368	-.369
.457	-.232	.170	-.481	-.191
.480	-.297	.200	-.456	-.006
.502	-.368	.235	-.487	.287
.551	-.407	.280	-.706	.414
.585	-.413	.312	-.868	.388
.592	-.387	.343	-1.018	-1.038
.613	-.310	.281	-.868	-.809
.634	-.239	.200	-.643	-.630
.655	-.194	.131	-.437	-.140
.675	-.116	.019	-.300	-.051
.696	-.026	.012	-.212	-.025
.774	.000	.056	-.094	.038
.852	.026	-.019	.019	-.115
.930	.103	-.287	.106	-.363

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.962	.899	.841	.862	.826
	.080	.475	.350	.318	.306	.355
	.130	-.368	-.593	-.681	-.606	-.581
	.145	-4.396	-4.289	-4.024	-4.358	-3.943
	.155	-1.630	-1.698	-1.668	-1.517	-1.239
	.180	-1.236	-1.099	-1.191	-1.005	-1.091
	.220	-.762	-.812	-.853	-.855	-.697
	.270	-.618	-.731	-.751	-.581	-.490
	.400	-.674	-.768	-.771	-.568	-.400
	.620	-1.230	-1.380	-1.146	-.231	-.071
	.685	-6.718	-6.631	-5.247		
	.693	-6.668	-7.305	-6.985		
	.700	-4.264	-5.239	-5.451		
	.720	-2.117	-2.323	-2.509	.169	.297
	.750	-1.474	-1.455	-1.929	-.031	.090
	.800	-.987	-.880	-1.668	-.194	-.019
	.900	-.568	-.318	-1.445	-.187	-.129
	.980	[illegible]	.156	-.618	-.181	-.181
Lower	.025	-.524	-.306	-.006	.031	-.219
	.120	-.425	-.256	-.051	-.056	-.284
	.220	-.337	-.343	-.083	-.106	-.200
	.300	.019	-.462	-.140	-.219	-.245
	.620	.687	.699	.287	-.493	-.336
	.750	.812	.843	-.178	-.506	-.348
	.850	.656	.743	.357	-.418	-.303
	.950	.524	.568	.248	-.318	-.258

α = 6.0°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.093	.494	.136	.359
.053	-.113	.257	-.052	.122
.100	-.199	.105	-.200	-.083
.145	-.139	.000	-.136	-.058
.189	-.066	.053	-.084	-.019
.234	-.099	.105	-.045	-.032
.280	-.106	.132	.057	-.045
.326	-.099	.145	-.019	-.013
.371	-.212	.198	-.194	.019
.392	-.245	.235	-.226	.186
.413	-.285	.277	-.516	.077
.434	-.371	.323	-.826	.173
.457	-.385	.345	-.755	.346
.480	-.444	.370	-.691	.429
.502	-.537	.395	-.703	.442
.551	-.497	.425	-.865	.462
.585	-.484	.441	-1.000	.513
.592	-.458	.408	-1.097	-1.071
.613	-.332	.316	-.871	-.630
.634	-.292	.224	-.613	-.590
.655	-.225	.119	-.394	-.397
.675	-.139	.000	-.252	-.109
.696	-.066	.000	-.161	.019
.774	-.033	.079	-.032	.058
.852	[illegible]	.040	.019	-.135
.930	.066	-.184	.084	-.212

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.701	.672	.647	.665	.696
	.080	-.107	-.184	-.154	-.200	-.099
	.130	-1.282	-1.488	-1.519	-1.426	-1.406
	.145	-6.691	-6.369	-5.737	-6.234	-5.650
	.155	-2.711	-2.766	-2.615	-2.497	-2.062
	.180	-2.003	-1.791	-1.795	-1.646	-1.664
	.220	-1.262	-1.311	-1.288	-1.297	-1.094
	.270	-.975	-1.113	-1.090	-.942	-.802
	.400	-.861	-1.014	-1.006	-.800	-.670
	.620	-1.115	-1.383	-1.301	-.336	-.239
	.685	-4.367	-6.500	-6.212		
	.693	-3.966	-7.462	-7.942		
	.700	-2.270	-5.335	-6.250		
	.720	-1.022	-2.358	-3.083	.006	.133
	.750	-.868	-1.455	-2.500	-.194	-.040
	.800	-.781	-.902	-2.186	-.323	-.106
	.900	-.668	-.375	-1.756	-.355	-.192
	.980	-.487	.066	-.615	-.290	-.159
Lower	.025	-.641	.349	.250	.161	-.119
	.120	.254	.237	.147	-.097	-.225
	.220	.507	.277	.154	-.045	-.093
	.300	.588	.435	.378	.277	.133
	.620	.714	.771	.429	-.142	-.166
	.750	.835	.896	-.526	-.981	-.504
	.850	.628	.724	.340	-.994	-.351
	.950	.387	.520	.244	-.691	-.252

α = 13.5°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.053	.588	-.115	.303
.053	-.220	.381	-.284	.101
.100	-.167	.160	-.372	-.182
.145	-.134	.073	-.332	-.161
.189	-.067	.107	-.278	-.108
.234	-.073	.160	-.102	-.141
.280	-.093	.200	.061	-.148
.326	-.120	.214	-.014	-.208
.371	-.274	.280	-.291	-.208
.392	-.320	.320	-.616	-.148
.413	-.347	.387	-.948	.208
.434	-.427	.441	-1.239	.531
.457	-.414	.450	-1.056	.619
.480	-.441	.460	-.914	.578
.502	-.501	.470	-.846	.551
.551	-.441	.480	-.928	.585
.585	-.414	.487	-.941	.652
.592	-.367	.394	-.975	-.948
.613	-.267	.314	-.779	-.525
.634	-.240	.240	-.535	-.410
.655	-.200	.114	-.345	-.410
.675	-.107	-.027	-.196	-.229
.696	-.053	.033	-.115	-.040
.774	.020	.147	.034	.101
.852	-.033	.060	-.014	-.034
.930	.053	-.040	.034	-.040

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.132	-1.048	-1.150	-.691	.040
	.080	-.935	-1.015	-1.103	-1.124	-.755
	.130	-2.384	-2.698	-2.858	-2.675	-2.411
	.145	-8.575	-8.694	-8.150	-8.749	-7.699
	.155	-3.602	-3.986	-4.034	-3.866	-3.058
	.180	-2.457	-2.577	-2.743	-2.492	-2.324
	.220	-1.561	-1.856	-1.923	-1.876	-1.522
	.270	-1.192	-1.509	-1.573	-1.361	-1.149
	.400	-.988	-1.235	-1.291	-1.070	-.875
	.620	-.929	-1.342	-1.365	-.488	-.361
	.685	-2.555	-5.756	-6.462		
	.693	-2.371	-7.118	-8.755		
	.700	-1.442	-5.081	-6.973		
	.720	-.718	-2.170	-3.550	-.122	.100
	.750	-.632	-1.309	-2.864	-.345	-.114
	.800	-.514	-.808	-2.501	-.440	-.214
	.900	-.468	-.421	-1.910	-.494	-.327
	.980	-.356	-.027	-.538	-.400	-.260
Lower	.025	.408	.661	.652	.596	.354
	.120	.790	.788	.726	.664	.548
	.220	.757	.728	.706	.650	.561
	.300	.659	.661	.612	.488	.361
	.620	.738	.768	.457	-.183	-.187
	.750	.843	.821	-.457	-1.138	-.608
	.850	.626	.708	.390	-1.077	-.427
	.950	.369	.494	.289	-.765	-.321

TABLE 12 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = -14°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.423	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 19.1°											
.032	-.187	.743	-.302	.169	Upper	.010	-4.103	-2.891	-3.081	-2.953	-2.370
.053	-.314	.518	-.467			.080	-1.761	-1.916	-2.560	-2.040	-1.349
.102	-.194	.296	-.522	-.284		.130	-3.176	-3.151	-3.311	-3.310	-3.392
.145	-.167	.198	-.495	-.298		.145	-9.852	-9.213	-8.661	-9.615	-9.121
.189	-.114	.225	-.440	-.257		.155	-4.246	-4.535	-4.584	-4.581	-3.926
.234	-.073	.280	-.158	-.311		.180	-2.797	-3.028	-3.203	-3.056	-2.931
.280	-.134	.286	.096	-.332		.220	-1.740	-2.175	-2.289	-2.266	-1.983
.326	-.167	.322	.048	-.433		.270	-1.293	-1.739	-1.842	-1.703	-1.509
.371	-.341	.389	-.330	-.460		.400	-.989	-1.323	-1.429	-1.250	-1.182
.392	-.400	.450	-.879	-.555		.620	-.921	-1.459	-1.578	-.694	-.634
.413	-.467	.535	-1.140	.176		.685	-2.925	-4.869	-5.884		
.434	-.541	.539	-1.738	.623		.693	-2.837	-5.518	-7.740		
.457	-.521	.545	-1.367	.691		.700	-1.767	-3.987	-6.196		
.480	[illegible]	.530	-1.120	.664		.720	-.806	-1.712	-3.257	-.364	-.114
.502	-.548	.525	-1.044	.616		.750	-.576	-1.002	-2.661	-.474	-.287
.551	-.421	.520	-1.016	.589		.800	-.535	-.515	-2.296	-.549	-.407
.585	-.374	.511	-1.181	.670		.900	-.427	-.215	-1.618	-.563	-.487
.592	-.334	.498	-1.243	-1.199		.980	-.352	-.061	-.454	-.481	-.387
.613	-.247	.396	-.907	-.400	Lower	.025	.670	.805	.792	.735	.548
.634	[illegible]	.280	-.556	-.372		.120	.867	.812	.738	.673	.501
.655	-.182	.136	-.343	-.345		.220	.826	.764	.765	.673	.507
.675	[illegible]	.020	-.179	-.210		.300	.738	.716	.650	.543	.354
.696	-.055	.034	-.082	-.054		.620	.792	.784	.494	-.110	-.247
.774	-.033	.170	.041	.122		.750	.874	.846	-.420	-1.126	-.594
.852	-.073	.075	-.069	.007		.850	.670	.716	.372	-1.058	-.427
.930	-.140	.061	-.014	.102		.950	.467	.418	.332	-.776	-.367
α = 22.9°											
.032	-.223	.797	-.444	.053	Upper	.010	-9.090	[illegible]	-4.200	-4.122	-4.002
.053	-.352	.591	-.551	-.087		.080	-2.179	[illegible]	-4.294	-4.095	-2.221
.102	-.212	.391	-.598	-.361		.130	-3.507	[illegible]	-3.399	-3.497	-3.657
.145	-.176	.275	-.558	-.401		.145	-10.044	[illegible]	-7.679	-8.089	-9.480
.189	-.281	.309	-.518	-.334		.155	-4.296	[illegible]	-4.454	-4.364	-4.320
.234	-.054	.323	-.182	-.407		.180	-2.857	[illegible]	-3.272	-3.113	-3.304
.280	-.115	.350	.134	-.454		.220	-1.805	[illegible]	-2.377	-2.306	-2.262
.326	-.223	.371	-.027	-.568		.270	-1.550	[illegible]	-1.936	-1.815	-1.781
.371	-.426	.460	-.444	-.654		.400	-1.128	[illegible]	-1.469	-1.338	-1.422
.392	-.500	.520	-.988	-.848		.620	-.996	[illegible]	-1.469	-.834	-.867
.413	-.596	.584	-1.278	.160		.685	-3.535	[illegible]	-3.532		
.434	-.622	.611	-2.031	.654		.693	-3.597	[illegible]	-5.041		
.457	-.569	.600	-1.547	.741		.700	-2.338	[illegible]	-4.046		
.480	-.515	.580	-1.331	.688		.720	-1.162	[illegible]	-2.190	-.471	-.257
.502	-.481	.570	-1.177	.668		.750	-.761	[illegible]	-1.830	-.538	-.467
.551	[illegible]	.560	-1.069	.641		.800	-.519	[illegible]	-1.556	-.592	-.576
.585	[illegible]	.549	-1.163	.661		.900	-.387	[illegible]	-1.128	-.545	-.616
.592	[illegible]	.522	-1.499	-1.436		.980	-.346	[illegible]	-.501	-.451	-.454
.613	-.156	.412	-1.190	-.425	Lower	.025	.782	[illegible]	.815	.773	.616
.634	[illegible]	.295	-.672	-.387		.120	.913	[illegible]	.721	.693	.488
.655	-.102	.172	-.370	-.287		.220	.851	[illegible]	.761	.706	.521
.675	-.054	.062	-.168	-.100		.300	.802	[illegible]	.661	.605	.420
.696	-.020	.076	-.061	.027		.620	.830	[illegible]	.541	.007	-.183
.774	-.014	.220	.047	.147		.750	.892	[illegible]	-.247	-.760	-.494
.852	-.095	.096	-.067	.040		.850	.733	[illegible]	.441	-.793	-.379
.930	-.120	.137	-.020	.147		.950	.526	[illegible]	.334	-.578	-.379

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TABLE 12 Continued
(b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 15°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.5°											
.032	.264	.286	.276	.289	Upper	.010	.943	.864	.861	.827	.779
.053	.050	.078	.045	.063		.080	.443	.299	.289	.269	.302
.100	-.113	-.065	-.122	-.094		.130	-.380	-.702	-.767	-.699	-.691
.145	-.075	-.091	-.090	-.031		.145	-4.415	-4.541	-4.204	-4.654	-4.098
.189	-.025	-.045	-.006	.013		.155	-1.632	-1.858	-1.791	-1.718	-1.339
.234	-.063	.032	-.032	.031		.180	-1.240	-1.228	-1.288	-1.147	-1.175
.280	-.063	.039	-.038	.025		.220	-.765	-.936	-.943	-.974	-.779
.326	-.044	.065	-.032	.075		.270	-.620	-.845	-.848	-.744	-.616
.371	-.113	.104	-.103	.132		.400	-.690	-.890	-.892	-.782	-.591
.392	-.142	.108	-.013	.327		.620	-1.246	-1.559	-1.345	-.942	-.641
.413	-.170	.117	-.179	-.220		.685	-6.680	-6.913	-5.386		
.434	-.226	.156	-.417	-.283		.693	-6.630	-7.543	-6.875	-2.083	-1.489
.457	-.245	.190	-.519	-.082		.700	-4.245	-5.444	-5.210	-2.122	-1.458
.480	-.302	.230	-.487	.031		.720	-2.100	-2.417	-2.149	-1.308	-.867
.502	-.402	.270	-.551	.220		.750	-1.455	-1.488	-1.364	-.962	-.660
.551	-.434	.310	-.737	.377		.800	-.981	-.884	-.880	-.705	-.553
.585	-.434	.338	-.936	.371		.900	-.588	-.299	-.704	-.385	-.440
.592	-.396	.331	-1.077	-1.056		.980	-.025	.156	-.295	-.205	-.365
.613	-.321	.273	-.917	-.742							
.634	-.258	.195	-.673	-.691	Lower	.025	-.392	-.234	.063	.115	-.113
.655	-.201	.130	-.462	-.170		.120	-.342	-.208	.025	.051	-.170
.675	-.126	.039	-.321	-.057		.220	-.234	-.247	.019	-.013	-.107
.696	-.063	[illegible]	-.244	-.031		.300	.000	-.312	-.057	-.077	-.138
.774	-.063	.045	-.096	.019		.620	.633	.643	.119	-.321	-.207
.852	-.025	-.032	-.013	-.132		.750	.816	.819	.006	-.147	-.101
.930	.063	-.279	.090	-.314		.850	.683	.767	.490	.019	-.113
						.950	.538	.572	.358	.045	-.163
α = 5.9°											
.032	.091	.478	.122	.356	Upper	.010	.607	.555	.540	.622	.611
.053	-.123	.232	-.096	.112		.080	-.155	-.245	-.310	-.288	-.201
.100	-.182	.071	-.205	-.092		.130	-1.329	-1.581	-1.752	-1.641	-1.611
.145	-.130		-.186	-.066		.145	-6.602	-6.492	-6.237	-6.699	-6.218
.189	-.065	.052	-.109	-.033		.155	-2.723	-2.859	-2.924	-2.801	-2.423
.234	-.097	.123	-.064	-.040		.180	-1.962	-1.852	-2.055	-1.846	-1.936
.280	-.091	.129	.058	-.033		.220	-1.245	-1.355	-1.488	-1.519	-1.332
.326	-.104	.148	.006	-.026		.270	-.962	-1.174	-1.278	-1.135	-1.072
.371	-.195	.219	-.167	.037		.400	-.865	-1.065	-1.212	-1.103	-1.001
.392	-.240	.245	-.276	.171		.620	-1.084	-1.529	-1.607	-1.212	-1.085
.413	-.286	.271	-.538	.184		.685	-3.904	-6.531	-6.105		
.434	-.344	.310	-.840	.217		.693	-3.485	-7.170	-7.686	-2.532	-2.579
.457	-.377	.335	-.782	.362		.700	-1.968	-5.072	-5.861	-2.609	-2.748
.480	-.422	.360	-.737	.454		.720	-.897	-2.213	-2.496	-1.667	-1.663
.502	-.520	.385	-.686	.468		.750	-.774	-1.329	-1.620	-1.276	-1.286
.551	-.474	.410	-.769	.507		.800	-.671	-.755	-1.054	-1.013	-1.052
.585	-.461	.432	-.853	.560		.900	-.568	-.258	-.731	-.590	-.741
.592	-.429	.426	-.853	-.863		.980	-.445	.142	-.277	-.282	-.364
.613	-.325	.336	-.679	-.620							
.634	-.279	.232	-.526	-.540	Lower	.025	.077	.381	.336	.282	.026
.655	-.227	.116	-.385	-.454		.120	.161	.329	.237	.064	-.058
.675	-.130	-.006	-.237	-.151		.220	.510	.374	.303	.135	.052
.696	-.078	.013	-.141	-.013		.300	.600	.529	.560	.423	.325
.774	-.045	.071	-.019	.007		.620	.703	.716	.580	.340	.065
.852	-.052	.039	-.026	-.158		.750	.813	.787	.382	.212	.253
.930	.058	-.142	.032	-.158		.850	.607	.697	.494	-.019	.104
						.950	.387	.529	.349	-.122	-.078
α = 13.3°											
.032	-.085	.630	-.102	.287	Upper	.010	-.167	-1.353	-1.676	-1.625	-.589
.053	-.255	.418	-.278	.073		.080	-.995	-1.061	-1.222	-1.341	-1.073
.100	-.196	.212	-.366	-.180		.130	-2.471	-2.719	-3.112	-3.074	-2.924
.145	-.144	.093	-.332	-.187		.145	-8.781	-8.687	-8.500	-9.453	-8.471
.189	-.098	.153	-.244	-.140		.155	-3.693	-4.005	-4.280	-4.334	-3.604
.234	-.111	.199	-.081	-.160		.180	-2.504	-2.606	-2.958	-2.891	-2.747
.280	-.118	.192	.074	-.187		.220	-1.609	-1.870	-2.123	-2.207	-1.884
.326	-.137	.239	.027	-.234		.270	-1.209	-1.525	-1.743	-1.693	-1.518
.371	-.268	.318	-.264	-.254		.400	-1.022	-1.220	-1.476	-1.463	-1.367
.392	-.318	.370	-.650	-.200		.620	-1.075	-1.565	-1.763	-1.537	-1.524
.413	-.379	.424	-.975	.207		.685	-2.684	-5.690	-6.103		
.434	-.432	.458	-1.300	.526		.693	-2.264	-6.359	-7.612	-2.932	-3.434
.457	-.445	.462	-1.090	.621		.700	-1.376	-4.456	-5.809	-2.993	-3.663
.480	-.464	.466	-.975	.568		.720	-.661	-1.863	-2.557	-1.998	-2.302
.502	-.523	.470	-.894	.528		.750	-.608	-1.028	-1.656	-1.537	-1.858
.551	-.471	.476	-.962	.568		.800	-.481	-.511	-1.102	-1.205	-1.518
.585	-.432	.484	-.955	.634		.900	-.394	-.153	-.721	-.718	-1.079
.592	-.399	.484	-.962	-.915		.980	-.361	.046	-.220	-.413	-.510
.613	-.301	.371	-.738	-.425							
.634	-.262	.259	-.508	-.381	Lower	.025	.494	.710	.688	.670	.471
.655	-.229	.119	-.345	-.394		.120	.848	.816	.728	.677	.523
.675	-.144	.007	-.210	-.267		.220	.795	.769	.741	.684	.543
.696	-.092	.033	-.129	-.080		.300	.694	.683	.654	.582	.392
.774	-.033	.153	.027	.087		.620	.808	.802	.588	.359	-.137
.852	-.085	.273	-.068	-.047		.750	.861	.875	.307	.176	.196
.930	[illegible]	[illegible]	-.007	[illegible]		.850	.648	.729	.487	-.027	.059
						.950	.394	.537	.367	-.129	-.092

TABLE 12 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 18°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.00

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 18.9°											
.032	-.184	.769	-.307	.136	Upper	.010	-4.415	-3.12	-3.464	-3.405	-3.194
.053	-.323	.563	-.427	-.041		.080	-1.740	-2.11	-3.103	-3.105	-1.699
.100	-.204	.330	-.514	-.293		.130	-3.114	-3.15	-3.382	-3.339	-3.629
.145	-.158	.234	-.481	-.314		.145	-9.537	-9.43	-8.845	-9.368	-9.582
.189	-.086	.240	-.414	-.273		.155	-4.095	-4.67	-4.780	-4.754	-4.373
.234	-.092	.288	-.160	-.314		.180	-2.721	-3.13	-3.403	-3.292	-3.339
.280	-.119	.309	.127	-.334		.220	-1.688	-2.24	-2.482	-2.484	-2.338
.326	-.138	.343	.013	-.423		.270	-1.269	-1.80	-2.032	-1.956	-1.903
.371	-.323	.412	-.300	-.491		.400	-.942	-1.37	-1.603	-1.576	-1.679
.392	.395	.475	-.868	-.586		.620	-.948	-1.67	-1.848	-1.656	-1.818
.413	-.468	.536	-1.108	.184		.685	-2.806	-4.80	-5.115		
.434	-.527	.577	-1.736	.627		.693	-2.662	-5.206	-6.513	-2.731	-3.984
.457	-.507	.570	-1.335	.702		.700	-1.622	-3.709	-4.978	-2.851	-4.287
.480	-.481	.564	-1.082	.655		.720	-.733	-1.566	-2.169	-1.976	-2.825
.502	-.514	.558	-1.002	.641		.750	-.523	-.831	-1.418	-1.583	-2.318
.551	-.402	.552	-.962	.607		.800	-.458	-.39	-.873	-1.249	-1.910
.585	-.356	.549	-1.082	.641		.900	-.340	-.13	-.552	-.775	-1.337
.592	-.329	.529	-1.175	-1.187		.980	-.314	-.02	-.157	-.521	-.639
.613	-.231	.398	-.868	-.425	Lower	.025	.693	.83	.791	.761	.580
.634	-.198	.275	-.561	-.327		.120	.870	.85	.730	.668	.494
.655	-.165	.144	-.347	-.321		.220	.837	.83	.764	.694	.527
.675	-.092	.027	-.167	-.232		.300	.746	.78	.689	.634	.428
.696	-.033	.062	-.067	-.061		.620	.778	.83	.627	.387	-.105
.774	-.013	.192	.040	.109		.750	.857	.88	.409	.214	.204
.852	-.072	.103	-.067	.007		.850	.641	.76	.532	.114	.105
.930	[illegible]	.096	-.007	.095		.950	.425	.56	.409	-.053	-.092
α = 22.9°											
.032	-.257	.817	-.457	.040	Upper	.010	-8.783	-4.08	-4.320	-4.209	-4.225
.053	-.366	.625	-.558	-.107		.080	-2.092	-3.90	-4.460	-4.223	-3.088
.100	-.203	.391	-.625	-.367		.130	-3.406	-3.09	-3.359	-3.530	-3.501
.145	-.183	.295	-.598	-.401		.145	-9.737	-8.21	-7.432	-7.269	-9.290
.189	-.095	.316	-.538	-.367		.155	-4.144	-4.46	-4.367	-4.075	-4.422
.234	-.068	.350	-.175	-.441		.180	-2.763	-3.13	-3.245	-2.959	-3.453
.280	-.115	.357	.134	-.454		.220	-1.747	-2.26	-2.364	-2.165	-2.465
.326	-.244	.371	.013	-.561		.270	-1.463	-1.80	-1.903	-1.681	-2.004
.371	-.440	.488	-.377	-.641		.400	-1.077	-1.33	-1.389	-1.224	-1.747
.392	-.520	.550	-.995	-.855		.620	-1.043	-1.40	-1.255	-.975	-1.652
.413	-.603	.604	-1.311	.174		.685	-3.704	-3.24	-1.048		
.434	-.650	.604	-2.044	.654		.693	-3.684	-3.61	-2.197	-1.136	-3.095
.457	-.582	.590	-1.587	.721		.700	-2.390	-2.56	-1.689	-1.210	-3.250
.480	-.508	.580	-1.372	.694		.720	-1.192	-1.09	-.948	-.995	-1.950
.502	-.488	.570	-1.170	.654		.750	-.792	-.68	-.895	-.901	-1.490
.551	-.311	.560	-1.069	.634		.800	-.535	-.48	-.835	-.908	-1.314
.585	-.257	.556	-1.123	.641		.900	-.393	-.37	-.688	-.894	-1.192
.592	-.230	.536	-1.499	-1.442		.980	-.440	-.30	-.668	-.881	-1.043
.613	-.169	.440	-1.170	-.430	Lower	.025	.792	.87	.788	.780	.596
.634	-.122	.302	-.659	-.387		.120	.914	.83	.728	.713	.481
.655	-.122	.158	-.363	-.307		.220	.860	.83	.775	.740	.555
.675	-.047	.014	-.175	-.100		.300	.772	.77	.694	.659	.447
.696	-.014	.062	-.054	.027		.620	.792	.83	.628	.403	[illegible]
.774	-.014	.206	.054	.140		.750	.880	.89	.421	.256	.244
.852	-.095	.089	-.027	.040		.850	.691	.72	.514	.101	.095
.930	-.014	.130	-.013	.127		.950	.488	.49	.307	-.215	-.203

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TABLE 12 Continued
(c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 30°; $\delta_{a,R}$ = 00°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
α = −1.5°											
.032	.282	.293	.277	.284	Upper	.010	.891	.860	.813	.845	.833
.053	.064	.076	.045	.052		.080	.385	.312	.245	.252	.308
.100	−.115	−.032	−.110	−.097		.130	−.468	−.669	−.852	−.794	−.705
.145	−.083	−.083	−.077	−.058		.145	−4.647	−4.394	−4.362	−4.834	−4.250
.189	−.026	−.032	−.013	−.006		.155	−1.763	−1.783	−1.871	−1.800	−1.429
.234	−.051	.038	−.052	.000		.180	−1.327	−1.172	−1.355	−1.220	−1.276
.280	−.051	.051	−.019	.013		.220	−.840	−.879	−1.007	−1.039	−.827
.326	−.032	.064	−.045	.052		.270	−.679	−.802	−.891	−.794	−.654
.371	−.109	.115	−.148	.129		.400	−.744	−.860	−.968	−.845	−.635
.392	−.136	.120	−.006	.323		.620	−1.308	−1.477	−1.426	−1.129	−.833
.413	−.167	.127	−.174	−.142		.685	−6.769	−6.648	−5.453	−2.478	−2.115
.434	−.199	.166	−.400	−.258		.693	−6.718	−7.081	−6.937	−3.556	−2.455
.457	−.237	.190	−.523	−.116		.700	−4.340	−5.209	−5.188	−2.562	−1.865
.480	−.308	.210	−.478	.052		.720	−2.115	−2.280	−2.091	−1.497	−1.135
.502	−.417	.240	−.536	.277		.750	−1.494	−1.407	−1.258	−1.097	−.904
.551	−.436	.275	−.742	.348		.800	−.981	−.809	−.723	−.832	−.724
.585	−.423	.312	−.916	.316		.900	−.603	−.242	−.413	−.574	−.590
.592	−.410	.337	−1.065	−1.084		.980	−.006	.197	−.194	−.323	−.494
.613	−.314	.267	−.897	−.761	Lower	.025	−.308	−.121	.090	.136	−.064
.634	−.250	.204	−.639	−.749		.120	−.276	−.089	.058	.065	−.122
.655	−.192	.127	−.458	−.187		.220	−.231	−.127	.013	.026	−.071
.675	−.122	.051	−.310	−.077		.300	.064	−.172	−.058	−.045	−.090
.696	−.045	−.006	−.213	−.019		.620	.615	.503	.155	−.097	−.141
.774	−.090	.038	−.084	.006		.750	.814	.713	.090	−.026	.032
.852	−.026	−.025	−.013	−.168		.850	.705	.764	.458	.084	.019
.930	.071	−.242	.103	−.323		.950	.538	.586	.329	.058	−.064
α = 5.8°											
.032	.113	.477	.140	.310	Upper	.010	.579	.523	.484	.568	.540
.053	−.107	.222	−.060	.116		.080	−.197	−.268	−.329	−.367	−.239
.100	−.182	.065	−.220	−.084		.130	−1.331	−1.622	−1.775	−1.796	−1.653
.145	−.132	−.013	−.160	−.077		.145	−6.597	−6.574	−6.266	−7.218	−6.247
.189	−.050	.052	−.100	−.026		.155	−2.751	−2.898	−2.956	−3.045	−2.451
.234	−.088	.085	−.020	−.032		.180	−1.987	−1.910	−2.078	−2.037	−1.998
.280	−.088	.111	.060	−.026		.220	−1.248	−1.400	−1.523	−1.669	−1.370
.326	−.101	.124	−.013	−.006		.270	−.955	−1.204	−1.297	−1.275	−1.106
.371	−.189	.177	−.180	.019		.400	−.834	−1.092	−1.271	−1.249	−1.068
.392	−.210	.225	−.260	.187		.620	−1.057	−1.603	−1.658	−1.583	−1.351
.413	−.245	.275	−.561	.226		.685	−3.572	−6.561	−6.163	−3.506	−3.865
.434	−.352	.301	−.855	.226		.693	−3.178	−7.234	−7.673	−5.188	−4.827
.457	−.358	.325	−.795	.368		.700	−1.808	−5.122	−5.827	−4.020	−4.135
.480	−.415	.345	−.748	.465		.720	−.828	−2.224	−2.446	−2.390	−2.363
.502	−.490	.365	−.708	.471		.750	−.739	−1.328	−1.523	−1.709	−1.810
.551	−.465	.390	−.808	.497		.800	−.643	−.759	−.910	−1.209	−1.395
.585	−.440	.419	−.881	.523		.900	−.529	−.235	−.407	−.661	−.993
.592	−.408	.392	−.861	−.826		.980	−.458	.124	−.097	−.314	−.484
.613	−.321	.314	−.694	−.620	Lower	.025	.146	.392	.381	.394	.132
.634	−.258	.216	−.528	−.529		.120	.197	.340	.316	.174	.025
.655	−.214	.072	−.401	−.419		.220	.503	.432	.407	.260	.189
.675	−.132	.000	−.254	−.161		.300	.605	.556	.607	.521	.415
.696	−.069	−.013	−.154	−.006		.620	.688	.706	.626	.507	.126
.774	−.063	.065	−.020	.013		.750	.783	.759	.549	.481	.415
.852	−.057	.026	−.027	−.148		.850	.599	.661	.561	.314	.264
.930	.044	−.150	.053	−.155		.950	.363	.497	.407	.120	−.006
α = 13.3°											
.032	−.058	.621	−.108	.269	Upper	.010	−.262	−1.378	−1.641	−1.625	−.936
.053	−.214	.409	−.311	.047		.080	−1.062	−1.071	−1.224	−1.348	−1.052
.100	−.169	.191	−.386	−.208		.130	−2.528	−2.789	−3.053	−3.027	−2.859
.145	−.104	.075	−.359	−.182		.145	−8.808	−8.940	−8.432	−9.372	−8.277
.189	−.065	.130	−.298	−.134		.155	−3.718	−4.126	−4.229	−4.300	−3.495
.234	−.084	.177	−.149	−.175		.180	−2.548	−2.700	−2.918	−2.844	−2.690
.280	−.117	.205	.054	−.175		.220	−1.654	−1.971	−2.098	−2.167	−1.819
.326	−.117	.225	−.027	−.222		.270	−1.251	−1.589	−1.701	−1.659	−1.442
.371	−.240	.286	−.264	−.242		.400	−1.062	−1.275	−1.425	−1.381	−1.286
.392	−.290	.345	−.643	−.175		.620	−1.143	−1.671	−1.560	−1.354	−1.507
.413	−.357	.416	−1.016	.195		.685	−2.871	−5.899	−4.680	−2.207	−4.509
.434	−.403	.464	−1.361	.551		.693	−2.474	−6.533	−5.998	−2.323	−5.717
.457	−.403	.466	−1.111	.598		.700	−1.573	−4.596	−4.424	−1.706	−4.892
.480	−.429	.468	−.955	.578		.720	−.706	−1.971	−1.782	−1.334	−2.709
.502	−.494	.470	−.901	.518		.750	−.625	−1.125	−1.183	−1.239	−2.092
.551	−.429	.473	−.948	.558		.800	−.498	−.586	−1.022	−1.300	−1.819
.585	−.396	.477	−.934	.646		.900	−.417	−.191	−.847	−1.327	−1.475
.592	−.370	.477	−.955	−.955		.980	−.498	−.123	−.659	−1.131	−.877
.613	−.292	.382	−.779	−.420	Lower	.025	.477	.682	.693	.677	.520
.634	−.247	.245	−.528	−.383		.120	.834	.798	.746	.691	.533
.655	−.208	.116	−.345	−.403		.220	.780	.764	.740	.684	.559
.675	−.117	−.007	−.203	−.242		.300	.652	.689	.646	.596	.455
.696	−.065	.027	−.122	−.074		.620	.753	.798	.659	.474	−.032
.774	.000	.150	.014	.087		.750	.854	.887	.511	.420	.377
.852	−.045	.041	−.047	−.061		.850	.632	.723	.511	.264	.247
.930	.039	−.007	.000	−.007		.950	.383	.532	.242	−.102	.000

TABLE 12 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 30°$; $\delta_{a,R} = 00°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.000$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = 19.0°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.192	.754	-.296	.144
.053	-.316	.533	-.430	-.041
.100	-.220	.318	-.524	-.302
.145	-.172	.214	-.471	-.309
.189	-.130	.247	-.417	-.254
.234	-.103	.292	-.161	-.316
.280	-.130	.292	.128	-.330
.326	-.158	.305	.020	-.433
.371	-.350	.390	-.309	-.467
.392	-.420	.440	-.854	-.556
.413	-.501	.507	-1.083	.179
.434	-.549	.539	-1.708	.625
.457	-.515	.535	-1.358	.707
.480	-.495	.531	-1.136	.659
.502	-.529	.527	-1.029	.646
.551	-.426	.523	-1.002	.611
.585	-.371	.520	-1.183	.666
.592	-.357	.500	-1.210	-1.250
.613	-.261	.416	-.861	-.425
.634	-.220	.260	-.572	-.385
.655	-.179	.162	-.336	-.357
.675	-.110	.032	-.188	-.234
.696	-.062	.045	-.087	-.055
.774	-.034	.182	.027	.124
.852	-.089	.065	-.067	-.014
.930	-.027	.097	-.034	.089

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-4.393	-2.911	-3.496	-3.463	-3.111
	.080	-1.712	-1.935	-3.159	-3.241	-1.635
	.130	-3.095	-2.935	-3.310	-3.322	-3.592
	.145	-9.523	-8.842	-8.723	-9.024	-9.540
	.155	-4.096	-4.365	-4.698	-4.613	-4.306
	.180	-2.740	-2.911	-3.324	-3.180	-3.297
	.220	-1.693	-2.079	-2.404	-2.374	-2.301
	.270	-1.271	-1.670	-1.937	-1.876	-1.848
	.400	-.935	-1.254	-1.504	-1.432	-1.641
	.620	-.988	-1.527	-1.607	-1.298	-1.751
	.685	-2.852	-4.489	-3.764	-1.506	-3.901
	.693	-2.674	-4.912	-4.986	-1.452	-4.629
	.700	-1.646	-3.495	-3.736	-1.177	-3.503
	.720	-.738	-1.442	-1.545	-1.015	-2.163
	.750	-.520	-.767	-1.133	-.988	-1.964
	.800	-.448	-.370	-1.010	-1.029	-1.655
	.900	-.382	-.143	-.907	-1.136	-1.532
	.980	-.435	-.162	-.776	-1.096	-1.367
Lower	.025	.692	.812	.824	.780	.625
	.120	.863	.825	.749	.699	.508
	.220	.817	.805	.797	.733	.543
	.300	.724	.754	.694	.652	.474
	.620	.771	.759	.714	.504	.048
	.750	.830	.845	.543	.451	.378
	.850	.659	.723	.556	.303	.206
	.950	.448	.507	.282	-.087	-.124

$\alpha = 22.9°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.251	.792	-.464	.027
.053	-.362	.609	-.573	-.102
.100	-.216	.406	-.600	-.355
.145	-.174	.291	-.580	-.443
.189	-.091	.325	-.511	-.375
.234	-.063	.345	-.205	-.443
.280	-.125	.379	.136	-.477
.326	-.216	.386	-.020	-.580
.371	-.418	.474	-.457	-.648
.392	-.500	.535	-1.009	-.852
.413	-.599	.596	-1.309	.157
.434	-.634	.616	-2.053	.661
.457	-.564	.600	-1.562	.730
.480	-.481	.585	-1.316	.696
.502	-.474	.560	-1.146	.655
.551	-.314	.548	-1.064	.600
.585	-.258	.542	-1.139	.634
.592	-.216	.515	-1.541	-1.493
.613	-.160	.406	-1.227	-.535
.634	-.111	.298	-.702	-.477
.655	-.111	.163	-.389	-.348
.675	-.042	.034	-.205	-.143
.696	.007	.068	-.089	-.014
.774	.000	.210	.034	.000
.852	-.091	.108	-.061	.020
.930	-.007	.142	-.020	.109

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-8.654	-3.961	-4.269	-3.976	-4.055
	.080	-2.100	-3.724	-4.378	-3.996	-2.439
	.130	-3.423	-2.966	-3.342	-3.348	-3.087
	.145	-9.929	-7.963	-7.317	-6.949	-8.117
	.155	-4.214	-4.280	-4.276	-3.833	-3.693
	.180	-2.796	-2.953	-3.164	-2.769	-2.892
	.220	-1.712	-2.140	-2.271	-2.025	-1.958
	.270	-1.453	-1.685	-1.821	-1.562	-1.568
	.400	-1.064	-1.219	-1.323	-1.166	-1.651
	.620	-1.030	-1.212	-1.166	-1.057	-1.401
	.685	-3.696	-2.770	-.614	-1.146	-1.923
	.693	-3.730	-3.183	-1.869	-1.282	-2.083
	.700	-2.455	-2.263	-1.534	-1.098	-1.826
	.720	-1.207	-1.002	-.948	-.975	-1.463
	.750	-.812	-.753	-.927	-.934	-1.373
	.800	-.566	-.651	-.873	-.955	-1.282
	.900	-.443	-.677	-.839	-.927	-1.205
	.980	-.477	-.453	-.743	-.921	-1.157
Lower	.025	.805	.874	.832	.798	.669
	.120	.927	.867	.777	.736	.571
	.220	.893	.842	.805	.771	.627
	.300	.791	.785	.750	.696	.564
	.620	.812	.833	.696	.552	.188
	.750	.893	.874	.573	.436	.439
	.850	.716	.725	.546	.280	.279
	.950	.484	.447	.245	-.089	-.091

L-927

L-726

TABLE 12 Continued (d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 37°$; $\delta_{a,R} = 00°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.000$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = -1.5°$											
.032	.274	.306	.274	.323	Upper	.010	.942	.866	.845	.828	.809
.053	.032	.083	.045	.071		.080	.442	.306	.265	.242	.299
.100	-.115	-.045	-.121	-.077		.130	-.391	-.681	-.839	-.745	-.669
.145	-.083	-.070	-.083	-.045		.145	-4.474	-4.438	-4.388	-4.757	-4.158
.189	-.045	-.019	-.013	.000		.155	-1.660	-1.815	-1.884	-1.751	-1.369
.234	-.051	.045	-.025	.006		.180	-1.269	-1.184	-1.375	-1.172	-1.210
.280	-.057	.064	-.032	.032		.220	-.776	-.904	-.987	-1.019	-.783
.326	-.051	.070	-.032	.084		.270	-.609	-.796	-.878	-.732	-.618
.371	-.115	.121	-.045	.148		.400	-.692	-.828	-.897	-.777	-.618
.392	-.144	.128	-.006	.361		.620	-1.224	-1.433	-1.239	-.904	-.802
.413	-.172	.134	-.166	-.058		.685	-6.615	-6.425	-4.401	-2.165	-2.834
.434	-.210	.159	-.427	-.181		.693	-6.571	-6.998	-5.685	-1.879	-2.566
.457	-.255	.183	-.522	-.110		.700	-4.224	-5.018	-4.149	-1.242	-1.732
.480	-.299	.207	-.484	.026		.720	-2.038	-2.197	-1.497	-1.153	-1.305
.502	-.408	.231	-.522	.168		.750	-1.404	-1.337	-.891	-1.089	-1.063
.551	-.433	.279	-.713	.303		.800	-.904	-.771	-.716	-1.032	-.802
.585	-.414	.312	-.891	.290		.900	-.558	-.242	-.523	-.866	-.675
.592	-.395	.318	-1.057	-1.097		.980	.013	.166	-.381	-.560	-.573
.613	-.312	.267	-.872	-.939	Lower	.025	-.237	-.089	.097	.134	-.051
.634	-.261	.197	-.624	-.781		.120	-.288	-.045	.071	.051	-.089
.655	-.204	.121	-.433	-.168		.220	-.192	-.108	.006	.025	-.057
.675	-.127	.038	-.293	-.058		.300	.019	-.115	-.065	-.025	-.108
.696	-.051	.013	-.204	-.026		.620	.596	.427	.110	-.051	-.178
.774	-.057	.045	-.076	.000		.750	.827	.637	.110	.038	.057
.852	-.019	-.038	.000	-.136		.850	.731	.745	.400	.153	.121
.930	.070	-.255	.096	-.303		.950	.571	.592	.336	.083	.045
$\alpha = 5.9°$											
.032	.097	.487	.133	.331	Upper	.010	.608	.561	.526	.610	.587
.053	-.103	.254	-.080	.097		.080	-.183	-.274	-.266	-.298	-.194
.100	-.200	.067	-.206	-.110		.130	-1.367	-1.629	-1.689	-1.678	-1.594
.145	-.123	-.020	-.166	-.084		.145	-6.750	-6.604	-6.081	-6.890	-6.034
.189	-.077	.033	-.080	-.065		.155	-2.773	-2.925	-2.833	-2.878	-2.343
.234	-.097	.100	-.053	-.045		.180	-2.021	-1.910	-1.988	-1.890	-1.846
.280	-.097	.100	.046	-.032		.220	-1.276	-1.389	-1.442	-1.519	-1.271
.326	-.097	.114	-.007	-.019		.270	-.975	-1.202	-1.228	-1.147	-1.013
.371	-.187	.180	-.166	.006		.400	-.870	-1.075	-1.137	-1.054	-.923
.392	-.232	.224	-.265	.175		.620	-1.125	-1.516	-1.345	-1.015	-.974
.413	-.277	.267	-.550	.195		.685	-3.977	-6.470	-4.489	-1.903	-2.852
.434	-.355	.300	-.862	.201		.693	-3.585	-7.212	-5.815	-1.545	-2.291
.457	-.368	.319	-.789	.351		.700	-2.015	-5.128	-4.249	-1.015	-1.362
.480	-.439	.338	-.696	.429		.720	-.935	-2.230	-1.572	-.968	-1.284
.502	-.510	.357	-.696	.468		.750	-.798	-1.356	-.981	-.975	-1.278
.551	-.484	.395	-.789	.487		.800	-.720	-.795	-.858	-1.028	-1.284
.585	-.452	.421	-.875	.539		.900	-.589	-.307	-.851	-1.081	-1.323
.592	-.426	.414	-.869	-.858		.980	-.543	-.073	-.669	-1.034	-1.129
.613	-.336	.347	-.696	-.708	Lower	.025	.078	.367	.351	.365	.148
.634	-.284	.220	-.531	-.552		.120	.170	.307	.266	.206	.026
.655	-.245	.100	-.378	-.435		.220	.517	.387	.318	.239	.136
.675	-.148	-.007	-.259	-.143		.300	.608	.554	.539	.484	.439
.696	-.097	.007	-.139	-.006		.620	.693	.721	.630	.570	.258
.774	-.065	.073	-.007	.019		.750	.785	.755	.617	.577	.510
.852	-.077	.040	.000	-.143		.850	.595	.661	.526	.365	.329
.930	.019	-.167	.053	-.175		.950	.360	.494	.195	-.040	-.052
$\alpha = 13.3°$											
.032	-.105	.621	-.078	.259	Upper	.010	-.232	-1.396	-1.605	-1.527	-.810
.053	-.263	.401	-.253	.040		.080	-1.028	-1.088	-1.220	-1.228	-1.034
.100	-.178	.227	-.364	-.192		.130	-2.480	-2.791	-3.050	-2.859	-2.825
.145	-.145	.107	-.305	-.186		.145	-8.667	-8.781	-8.349	-8.881	-8.232
.189	-.092	.134	-.260	-.119		.155	-3.674	-4.067	-4.178	-4.061	-3.451
.234	-.092	.187	-.123	-.166		.180	-2.500	-2.631	-2.871	-2.677	-2.641
.280	-.119	.214	.078	-.186		.220	-1.605	-1.903	-2.062	-2.034	-1.778
.326	-.125	.227	.026	-.232		.270	-1.227	-1.542	-1.691	-1.546	-1.409
.371	-.283	.300	-.240	-.245		.400	-1.041	-1.249	-1.406	-1.299	-1.264
.392	-.333	.354	-.604	-.192		.620	-1.074	-1.536	-1.545	-1.215	-1.357
.413	-.382	.407	-.929	.225		.685	-2.706	-5.636	-4.841	-2.085	-3.385
.434	-.428	.467	-1.241	.550		.693	-2.334	-6.357	-6.194	-1.455	-2.687
.457	-.428	.470	-1.027	.630		.700	-1.426	-4.501	-4.576	-1.065	-1.693
.480	-.474	.473	-.916	.597		.720	-.683	-1.916	-1.850	-1.052	-1.633
.502	-.527	.476	-.838	.524		.750	-.603	-1.102	-1.273	-1.052	-1.620
.551	-.461	.482	-.884	.584		.800	-.497	-.574	-1.174	-1.130	-1.554
.585	-.421	.487	-.884	.630		.900	-.438	-.214	-1.107	-1.286	-1.614
.592	-.395	.467	-.897	-.922		.980	-.564	-.254	-.935	-1.247	-1.508
.613	-.296	.354	-.708	-.600	Lower	.025	.471	.694	.683	.676	.527
.634	-.277	.260	-.507	-.391		.120	.802	.815	.749	.708	.540
.655	-.224	.120	-.325	-.431		.220	.776	.761	.749	.708	.599
.675	-.151	.013	-.208	-.272		.300	.656	.694	.643	.617	.461
.696	-.086	.027	-.110	-.086		.620	.749	.801	.710	.565	.112
.774	-.046	.160	.019	-.083		.750	.829	.895	.630	.565	.474
.852	-.059	.080	-.045	-.080		.850	.603	.721	.524	.390	.290
.930	.007	.020	.000	-.033		.950	.378	.534	.206	-.045	-.132

TABLE 12 Concluded
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 37°$; $\delta_{a,R} = 00°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.000$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = 19.0°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.153	.768	-.332	.178
.053	-.305	.561	-.470	-.026
.100	-.219	.300	-.560	-.250
.145	-.153	.207	-.519	-.290
.189	-.086	.254	-.463	-.237
.234	-.086	.307	-.180	-.303
.280	-.126	.300	.118	-.323
.326	-.153	.321	.014	-.408
.371	-.332	.387	-.339	-.487
.392	-.395	.464	-.920	-.566
.413	-.458	.541	-1.155	.204
.434	-.537	.561	-1.819	.612
.457	-.497	.555	-1.425	.692
.480	-.484	.548	-1.169	.665
.502	-.517	.542	-1.086	.626
.551	-.398	.530	-1.017	.593
.585	-.351	.521	-1.128	.652
.592	-.338	.514	-1.266	-1.166
.613	-.252	.401	-.948	-.600
.634	-.206	.274	-.616	-.362
.655	-.186	.147	-.374	-.336
.675	-.099	.040	-.221	-.244
.696	-.060	.067	-.118	-.099
.774	-.013	.077	.021	-.050
.852	-.099	.087	-.076	[illegible]
.930	.000	.107	-.035	.092

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-4.828	-3.058	-3.398	-3.652	-3.156
	.080	-1.868	-2.030	-3.062	-3.493	-1.684
	.130	-3.327	-3.072	-3.194	-3.431	-3.442
	.145	-10.176	-9.088	-8.357	-9.276	-9.284
	.155	-4.379	-4.514	-4.544	-4.773	-4.290
	.180	-2.905	-3.025	-3.227	-3.334	-3.229
	.220	-1.792	-2.163	-2.331	-2.504	-2.294
	.270	-1.390	-1.723	-1.890	-1.978	-1.857
	.400	-1.045	-1.329	-1.469	-1.563	-1.631
	.620	-1.072	-1.549	-1.567	-1.418	-1.452
	.685	-3.106	-4.614	-3.971	-1.833	-2.918
	.693	-2.940	-5.195	-5.209	-1.446	-2.142
	.700	-1.805	-3.653	-3.899	-1.128	-1.492
	.720	-.851	-1.549	-1.646	-1.114	-1.452
	.750	-.616	-.855	-1.172	-1.134	-1.452
	.800	-.498	-.414	-1.067	-1.204	-1.452
	.900	-.422	-.200	-1.067	-1.300	-1.519
	.980	-.602	-.341	-.863	-1.252	-1.446
Lower	.025	.713	.815	.803	.816	.637
	.120	.906	.828	.744	.706	.517
	.220	.837	.808	.784	.747	.577
	.300	.754	.761	.698	.678	.524
	.620	.823	.841	.718	.588	.153
	.750	.865	.881	.652	.602	.471
	.850	.685	.721	.566	.394	.305
	.950	.450	.514	.257	[illegible]	-.086

$\alpha = 22.9°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.263	.813	-.460	.055
.053	-.360	.609	-.577	-.096
.100	-.221	.400	-.666	-.364
.145	-.180	.325	-.618	-.426
.189	-.090	.332	-.536	-.350
.234	-.076	.366	-.213	-.426
.280	-.125	.379	.124	-.467
.326	-.228	.386	.062	-.591
.371	-.429	.488	-.426	-.680
.392	-.526	.542	-1.030	-.865
.413	-.623	.596	-1.339	.199
.434	-.650	.630	-2.115	.673
.457	-.581	.615	-1.607	.762
.480	-.498	.600	-1.387	.728
.502	-.491	.585	-1.209	.673
.551	-.318	.560	-1.078	.632
.585	-.256	.548	-1.140	.666
.592	-.235	.535	-1.490	-1.504
.613	-.152	.433	-1.284	-.520
.634	-.138	.291	-.776	-.433
.655	-.138	.156	-.426	-.337
.675	-.048	.034	-.247	-.124
.696	-.021	.054	-.137	.007
.774	-.055	.078	.034	.010
.852	-.111	.102	-.069	.021
.930	-.042	.142	-.034	.130

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-9.420	-4.022	-4.354	-4.169	-4.019
	.080	-2.174	-3.873	-4.485	-4.196	-2.435
	.130	-3.498	-3.000	-3.448	-3.606	-2.947
	.145	-10.054	-7.923	-7.315	-6.827	-7.623
	.155	-4.264	-4.313	-4.334	-3.915	-3.514
	.180	-2.801	-3.027	-3.221	-2.905	-2.670
	.220	-1.798	-2.201	-2.370	-2.163	-1.799
	.270	-1.526	-1.761	-1.916	-1.745	-1.425
	.400	-1.129	-1.300	-1.449	-1.394	-1.397
	.620	-1.052	-1.300	-1.339	-1.168	-1.300
	.685	-3.770	-3.196	-1.147	-1.566	-1.418
	.693	-3.902	-3.772	-2.335	-1.566	-1.300
	.700	-2.557	-2.681	-1.902	-1.243	-1.224
	.720	-1.268	-1.138	-1.133	-1.140	-1.190
	.750	-.850	-.738	-1.058	-1.092	-1.176
	.800	-.585	-.521	-.982	-1.051	-1.183
	.900	-.453	-.427	-.872	-1.037	-1.155
	.980	-.557	-.318	-.721	-.989	-1.079
Lower	.025	.822	.880	.838	.852	.664
	.120	.920	.846	.776	.755	.574
	.220	.885	.833	.817	.804	.650
	.300	.794	.799	.749	.742	.595
	.620	.808	.846	.749	.611	.228
	.750	.885	.887	.694	.639	.498
	.850	.690	.752	.556	.467	.353
	.950	.488	.488	.268	.048	[illegible]

TABLE 13 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 60°; $\delta_{a,R}$ = 30°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.8°											
.032	.229	.321	.277	.272	Upper	.010	.923	.863	.791	.820	.791
.053	.046	.072	.045	.063		.080	.357	.268	.196	.187	.249
.100	-.124	-.039	-.110	-.082		.130	-.507	-.798	-.886	-.884	-.857
.145	-.098	-.072	-.084	-.051		.145	-4.951	-4.801	-4.491	-5.163	-4.755
.189	-.020	-.026	-.019	-.019		.155	-1.865	-1.988	-1.942	-1.988	-1.589
.234	-.052	.059	.000	.000		.180	-1.410	-1.308	-1.417	-1.329	-1.413
.280	-.072	.072	-.032	.032		.220	-.877	-.988	-1.056	-1.149	-.968
.326	-.065	.098	-.026	.070		.270	-.708	-.896	-.962	-.878	-.791
.371	-.137	.137	-.097	.120		.400	-.747	-.935	-1.025	-1.020	-.850
.392	-.170	.155	-.006	.110		.620	-1.319	-1.589	-1.544	-1.607	-1.361
.413	-.203	.170	-.213	.095		.685	-6.900	-6.986	-5.801	-9.293	-6.476
.434	-.255	.170	-.452	-.057		.693	-6.809	-7.673	-7.332	-9.325	-5.004
.457	-.294	.200	-.581	-.006		.700	-4.359	-5.495	-5.541	-5.408	-3.683
.480	-.334	.230	-.523	.070		.720	-2.118	-2.401	-2.246	-2.343	-1.603
.502	-.438	.260	-.561	.164		.750	-1.429	-1.459	-1.335	-1.336	-1.334
.551	-.477	.290	-.749	.310		.800	-.994	-.857	-.765	-.716	-1.158
.585	-.464	.301	-.910	.259		.900	-.637	-.242	-.436	-.303	-1.001
.592	-.438	.301	-1.039	-.879		.980	-.026	.242	-.114	.032	-.844
.613	-.327	.242	-.858	-.677	Lower	.025	-.039	.039	.171	.181	.078
.634	-.275	.190	-.600	-.854		.120	-.123	.052	.158	.161	.039
.655	-.229	.124	-.394	-.171		.220	-.071	.033	.120	.136	.039
.675	-.124	.052	-.258	-.070		.300	.065	.026	.082	.065	-.026
.696	-.072	.026	-.181	-.025		.620	.468	.340	.215	.161	-.026
.774	-.046	.059	-.058	.000		.750	.728	.510	.323	.310	.340
.852	-.026	-.039	-.013	-.139		.850	.760	.648	.411	.419	.517
.930	.072	-.229	.077	-.266		.950	.591	.595	.392	.503	.445
α = 5.4°											
.032	.080	.474	.134	.321	Upper	.010	.531	.481	.367	.341	.461
.053	-.127	.254	-.093	.067		.080	-.265	-.347	-.514	-.588	-.381
.100	-.194	.053	-.227	-.120		.130	-1.485	-1.796	-2.123	-2.137	-1.970
.145	-.147	-.013	-.174	-.087		.145	-7.155	-7.118	-7.125	-8.033	-7.192
.189	-.073	.033	-.127	-.040		.155	-2.951	-3.165	-3.446	-3.499	-2.838
.234	-.114	.120	.000	-.047		.180	-2.175	-2.063	-2.444	-2.364	-2.330
.280	-.114	.120	.060	-.040		.220	-1.373	-1.549	-1.810	-1.950	-1.663
.326	-.114	.134	.047	-.040		.270	-1.068	-1.342	-1.569	-1.536	-1.376
.371	-.234	.234	-.160	.000		.400	-.948	-1.215	-1.516	-1.563	-1.429
.392	-.280	.265	-.307	.147		.620	-1.187	-1.796	-2.123	-2.397	-2.357
.413	-.327	.307	-.628	.267		.685	-3.886	-7.131	-7.679	-13.061	-6.476
.434	-.381	.354	-.975	.300		.693	-3.375	-7.859	-9.308	-13.208	-13.522
.457	-.434	.370	-.901	.421		.700	-1.950	-5.609	-7.145	-8.006	-10.023
.480	-.494	.390	-.808	.481		.720	-.935	-2.437	-3.118	-3.839	-4.708
.502	-.574	.410	-.768	.487		.750	-.796	-1.462	-1.910	-2.377	-3.332
.551	-.534	.430	-.855	.494		.800	-.690	-.855	-1.082	-1.349	-2.618
.585	-.521	.441	-.921	.561		.900	-.603	-.287	-.414	-.474	-1.896
.592	-.474	.421	-.888	-.868		.980	-.511	.127	.067	.160	-.841
.613	-.327	.334	-.708	-.620	Lower	.025	.206	.434	.387	.521	.180
.634	-.314	.240	-.561	-.581		.120	.292	.421	.434	.568	.220
.655	-.260	.120	-.401	-.521		.220	.537	.521	.721	.601	.568
.675	-.167	.020	-.247	-.214		.300	.603	.608	.681	.614	.514
.696	-.087	.027	-.140	-.060		.620	.683	.728	.761	.748	.013
.774	-.033	.114	.013	.027		.750	.782	.801	.795	.808	.574
.852	-.047	.067	-.013	-.127		.850	.590	.674	.668	.768	.528
.930	.033	-.107	.047	-.127		.950	.351	.548	.514	.628	.334
α = 13.0°											
.032	-.080	.645	-.122	.263	Upper	.010	-.323	-1.614	-2.028	-2.275	-2.037
.053	-.234	.402	-.284	.046		.080	-1.130	-1.100	-1.251	-1.449	-1.262
.100	-.194	.231	-.379	-.178		.130	-2.649	-2.885	-3.227	-3.291	-3.439
.145	-.140	.145	-.332	-.178		.145	-9.145	-8.990	-8.759	-10.008	-9.509
.189	-.087	.171	-.284	-.145		.155	-3.907	-4.235	-4.505	-4.781	-4.147
.234	-.114	.237	-.068	-.171		.180	-2.649	-2.753	-3.135	-3.223	-3.232
.280	-.134	.217	.074	-.184		.220	-1.694	-1.969	-2.285	-2.519	-2.290
.326	-.154	.244	.041	-.237		.270	-1.291	-1.620	-1.910	-1.998	-1.903
.371	-.300	.329	-.257	-.250		.400	-1.049	-1.330	-1.686	-1.849	-1.830
.392	-.345	.390	-.664	-.224		.620	-1.197	-1.857	-2.180	-2.709	-2.764
.413	-.394	.448	-1.036	.211		.685	-2.703	-6.046	-7.139	-13.462	-6.476
.434	-.461	.474	-1.395	.547		.693	-2.259	-6.428	-8.805	-13.529	-14.617
.457	-.467	.472	-1.138	.665		.700	-1.385	-4.584	-6.751	-8.288	-10.777
.480	-.507	.470	-1.002	.586		.720	-.646	-1.949	-2.964	-4.076	-5.001
.502	-.554	.468	-.934	.547		.750	-.558	-1.093	-1.818	-2.539	-3.546
.551	-.487	.465	-.962	.566		.800	-.457	-.547	-1.014	-1.503	-2.945
.585	-.467	.461	-.955	.626		.900	-.424	-.191	-.362	-.528	-2.384
.592	-.421	.461	-.955	-.876		.980	-.464	-.171	.013	-.244	-1.669
.613	-.287	.356	-.765	-.400	Lower	.025	.538	.731	.731	.752	.581
.634	-.287	.263	-.555	-.362		.120	.854	.830	.771	.718	.521
.655	-.240	.132	-.379	-.441		.220	.807	.777	.764	.745	.568
.675	-.140	-.007	-.230	-.283		.300	.699	.731	.711	.691	.501
.696	-.093	.046	-.108	-.099		.620	.767	.817	.810	.731	-.100
.774	-.020	.165	.027	.079		.750	.861	.889	.803	.779	.514
.852	-.053	.079	-.041	-.033		.850	.646	.738	.705	.731	.454
.930	.040	.026	.000	.033		.950	.390	.540	.527	.623	.220

TABLE 13 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 60°; $\delta_{a,R}$ = 30°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing , flap , or aileron				
					α = 18.7°						
.032	-.149	.771	-.303	.169	Upper	.010	-5.349	-3.410	-3.995	-4.176	-3.995
.053	-.311	.539	-.464	-.007		.080	-1.876	-2.489	-3.914	-4.176	-2.918
.100	-.190	.321	-.518	-.257		.130	-3.325	-3.171	-3.365	-3.638	-3.724
.145	-.163	.232	-.491	-.305		.145	-10.191	-9.465	-8.694	-9.145	-10.049
.189	-.081	.266	-.424	-.251		.155	-4.442	-4.808	-4.896	-4.983	-4.754
.234	-.095	.314	-.128	-.305		.180	-2.993	-3.226	-3.548	-3.577	-3.751
.280	-.135	.314	.121	-.345		.220	-1.828	-2.346	-2.607	-2.777	-2.736
.326	-.169	.327	.108	-.427		.270	-1.429	-1.903	-2.140	-2.253	-2.302
.371	-.339	.430	-.316	-.488		.400	-1.056	-1.446	-1.781	-1.943	-2.235
.392	-.420	.490	-.914	-.609		.620	-1.097	-1.875	-2.133	-2.575	-2.736
.413	-.508	.559	-1.190	.203		.685	-2.763	-5.026	-5.790	-9.272	-6.476
.434	-.562	.559	-1.849	.643		.693	-2.505	-5.401	-7.279	-9.232	-13.285
.457	-.528	.555	-1.446	.711		.700	-1.517	-3.819	-5.600	-5.568	-9.724
.480	-.535	.550	-1.150	.677		.720	-.718	-1.589	-2.451	-2.710	-4.442
.502	-.555	.545	-1.069	.677		.750	-.603	-.880	-1.510	-1.634	-3.169
.551	-.447	.540	-1.022	.630		.800	-.488	-.436	-.840	-.901	-2.587
.585	-.420	.539	-1.163	.677		.900	-.372	-.218	-.318	-.572	-2.018
.592	-.393	.511	-1.163	-1.111		.980	-.372	-.184	.007	-.430	-1.354
.613	-.257	.402	-.841	-.400	Lower	.025	.704	.859	.819	.800	.582
.634	-.237	.286	-.572	-.345		.120	.874	.852	.765	.719	.481
.655	-.217	.177	-.336	-.332		.220	.833	.825	.813	.760	.569
.675	-.108	.034	-.195	-.257		.300	.745	.777	.752	.693	.474
.696	-.068	.055	-.108	-.088		.620	.785	.832	.819	.760	-.115
.774	.007	.205	-.075	-.035		.750	.853	.887	.813	.767	.501
.852	-.081	.136	-.047	.020		.850	.637	.743	.691	.733	.460
.930	-.014	.136	-.020	.142		.950	.420	.525	.494	.565	.284
					α = 22.9°						
.032	-.260	.799	-.481	.027	Upper	.010	-9.013	-3.957	-4.307	-4.137	-3.953
.053	-.374	.617	-.569	-.093		.080	-2.140	-3.781	-4.414	-4.151	-2.591
.100	-.227	.403	-.623	-.134		.130	-3.494	-2.956	-3.372	-3.548	-2.531
.145	-.214	.299	-.569	-.387		.145	-10.056	-8.004	-7.238	-6.548	-6.577
.189	-.127	.318	-.535	-.327		.155	-4.293	-4.353	-4.280	-3.778	-2.898
.234	-.093	.364	-.129	-.414		.180	-2.817	-3.021	-3.158	-2.797	-2.317
.280	-.140	.364	.156	-.467		.220	-1.788	-2.183	-2.297	-2.092	-1.502
.326	-.247	.383	.041	-.588		.270	-1.503	-1.728	-1.850	-1.679	-1.169
.371	-.434	.474	-.481	-.641		.400	-1.097	-1.299	-1.409	-1.341	-1.128
.392	-.520	.525	-1.002	-.868		.620	-1.043	-1.377	-1.322	-1.043	-1.075
.413	-.628	.585	-1.314	.200		.685	-3.765	-3.541	-1.048	-1.937	-2.090
.434	-.668	.604	-2.072	.668		.693	-3.663	-3.924	-2.350	-2.065	-1.810
.457	-.594	.590	-1.585	.741		.700	-2.417	-2.820	-1.963	-1.476	-1.596
.480	-.534	.575	-1.334	.714		.720	-1.246	-1.215	-1.149	-1.104	-1.122
.502	-.548	.560	-1.199	.681		.750	-.867	-.747	-1.048	-1.083	-1.115
.551	-.374	.545	-1.097	.628		.800	-.603	-.487	-1.055	-1.029	-1.095
.585	-.314	.526	-1.151	.661		.900	-.494	-.318	-.968	-1.002	-1.008
.592	-.294	.507	-1.530	-1.422		.980	-.420	-.149	-.828	-.989	-.955
.613	-.167	.409	-1.273	-.550	Lower	.075	.785	.871	.868	.846	.661
.634	-.187	.292	-.785	-.494		.120	.707	.858	.801	.772	.588
.655	-.174	.162	-.440	-.347		.220	.867	.838	.841	.826	.661
.675	-.107	.026	-.251	-.127		.300	.799	.793	.761	.765	.574
.696	-.067	.071	-.108	-.013		.620	.792	.832	.801	.792	.247
.774	-.053	.201	-.090	.107		.750	.874	.877	.828	.792	.614
.852	-.154	.110	-.088	.013		.850	.697	.721	.581	.718	.554
.930	-.073	.162	-.014	.127		.950	.460	.494	.220	.400	.294

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TABLE 13 Continued
(b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 30°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = −1.8°

x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)
.032	.255	.300	.286	.288
.053	.013	.100	.065	.051
.100	-.096	-.019	-.091	-.109
.145	-.102	-.075	-.071	-.045
.189	-.019	-.019	-.013	[illegible]
.234	-.051	.056	.006	.006
.280	-.057	.075	-.013	.026
.326	-.057	.094	-.006	.058
.371	-.121	.137	-.110	.122
.392	-.150	.150	-.145	.040
.413	-.197	.169	-.182	-.051
.434	-.229	.200	-.435	-.199
.457	-.267	.220	-.546	-.077
.480	-.331	.240	-.507	.051
.502	-.414	.260	-.546	.179
.551	-.465	.280	-.741	.353
.585	-.452	.300	-.916	.308
.592	-.433	.325	-1.059	-1.038
.613	-.325	.268	-.890	-.737
.634	-.280	.200	-.617	-.840
.655	-.217	.131	-.403	-.179
.675	-.121	.050	-.273	-.064
.696	-.076	.037	-.175	-.038
.774	-.045	.075	-.071	.019
.852	-.025	-.012	.013	-.135
.930	.064	-.206	.110	-.288

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.918	.855	.853	.838	.815
	.080	.421	.287	.256	.221	.267
	.130	-.408	-.706	-.821	-.819	-.764
	.145	-4.613	-4.620	-4.423	-5.022	-4.419
	.155	-1.716	-1.867	-1.910	-1.897	-1.503
	.180	-1.313	-1.230	-1.372	-1.267	-1.331
	.220	-.798	-.949	-1.032	-1.111	-.885
	.270	-.641	-.849	-.923	-.838	-.720
	.400	-.698	-.874	-.987	-.955	-.751
	.620	-1.251	-1.542	-1.551	-1.397	-1.114
	.685	-6.756	-6.793	-6.051	-5.542	-5.158
	.693	-6.687	-7.349	-7.699	-6.835	-4.661
	.700	-4.267	-5.345	-5.872	-4.515	-3.630
	.720	-2.080	-2.348	-2.410	-2.163	-1.496
	.750	-1.408	-1.417	-1.442	-1.442	-1.121
	.800	-.943	-.837	-.801	-.910	-.942
	.900	-.566	-.225	-.295	-.468	-.771
	.980	.013	.250	[illegible]	-.104	-.554
Lower	.025	-.182	-.019	.135	.162	-.038
	.120	-.214	.006	.109	.097	-.089
	.220	-.170	-.012	.071	.058	-.051
	.300	.050	-.037	-.013	.026	-.083
	.620	.559	.412	.147	.045	-.134
	.750	.786	.631	.205	.143	.102
	.850	.723	.743	.391	.234	.146
	.950	.559	.618	.404	.234	.070

α = 5.5°

x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)
.032	.085	.465	.148	.314
.053	-.118	.223	-.090	.098
.100	-.190	.057	-.226	-.118
.145	-.150	[illegible]	-.181	-.092
.189	-.065	.045	-.110	-.033
.234	-.085	.108	-.026	-.039
.280	-.098	.127	.039	-.039
.326	-.111	.146	.013	-.033
.371	-.209	.217	-.148	-.007
.392	-.290	.255	-.323	.157
.413	-.321	.293	-.581	.242
.434	-.392	.344	-.891	.242
.457	-.412	.360	-.839	.353
.480	-.458	.380	-.761	.464
.502	-.549	.400	-.729	.484
.551	-.517	.420	-.800	.497
.585	-.491	.433	-.878	.549
.592	-.458	.420	-.852	-.778
.613	-.334	.337	-.691	-.625
.634	-.301	.217	-.516	-.536
.655	-.255	.134	-.387	-.477
.675	-.150	.025	-.245	-.196
.696	-.092	.013	-.148	-.013
.774	-.007	.096	-.019	.026
.852	-.033	.070	-.019	-.124
.930	.065	-.102	.065	-.124

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.546	.497	.425	.458	.543
	.080	-.247	-.331	-.412	-.452	-.275
	.130	-1.449	-1.675	-1.923	-1.846	-1.786
	.145	-6.978	-6.686	-6.593	-7.305	-6.672
	.155	-2.904	-2.974	-3.153	-3.117	-2.656
	.180	-2.124	-1.929	-2.224	-2.091	-2.119
	.220	-1.351	-1.439	-1.635	-1.710	-1.504
	.270	-1.040	-1.242	-1.426	-1.329	-1.230
	.400	-.923	-1.114	-1.361	-1.329	-1.236
	.620	-1.163	-1.662	-1.845	-1.794	-1.740
	.685	-3.736	-6.654	-6.829	-6.660	-8.471
	.693	-3.307	-7.291	-8.373	-8.228	-8.471
	.700	-1.878	-5.158	-6.378	-5.576	-6.757
	.720	-.858	-2.248	-2.721	-2.839	-3.205
	.750	-.754	-1.356	-1.681	-1.962	-2.361
	.800	-.650	-.777	-.981	-1.329	-1.890
	.900	-.552	-.248	-.366	-.697	-1.387
	.980	-.461	.096	.026	-.161	-.628
Lower	.025	.182	.408	.373	.413	.164
	.120	.247	.363	.360	.258	.052
	.220	.533	.471	.484	.400	.203
	.300	.611	.579	.615	.568	.438
	.620	.682	.700	.667	.574	.124
	.750	.799	.758	.661	.587	.471
	.850	.585	.681	.615	.478	.347
	.950	.370	.522	.471	.316	.150

α = 13.0°

x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)
.032	-.061	.632	-.089	.252
.053	-.259	.408	-.242	.046
.100	-.225	.191	-.344	-.192
.145	-.150	.099	-.318	-.206
.189	-.095	.151	-.274	-.146
.234	-.116	.217	-.076	-.192
.280	-.143	.231	.089	-.186
.326	-.150	.244	.038	-.239
.371	-.286	.323	-.204	-.239
.392	-.350	.385	-.624	-.239
.413	-.416	.441	-.955	.219
.434	-.471	.501	-1.299	.544
.457	-.450	.502	-1.089	.637
.480	-.511	.504	-.936	.597
.502	-.559	.506	-.860	.557
.551	-.511	.507	-.923	.590
.585	-.471	.507	-.898	.656
.592	-.423	.487	-.879	-.875
.613	-.314	.395	-.694	-.113
.634	-.307	.277	-.497	-.358
.655	-.245	.138	-.325	-.411
.675	-.157	.020	-.197	-.285
.696	-.109	.026	-.115	-.086
.774	-.020	.178	-.075	.106
.852	-.061	.092	-.032	-.040
.930	.007	.053	.013	.027

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.312	-1.488	-1.930	-1.847	-1.759
	.080	-1.088	-1.087	-1.260	-1.318	-1.289
	.130	-2.593	-2.832	-3.190	-3.057	-3.417
	.145	-8.939	-8.891	-8.727	-9.348	-9.540
	.155	-3.820	-4.129	-4.430	-4.387	-4.160
	.180	-2.593	-2.700	-3.077	-2.929	-3.198
	.220	-1.678	-1.949	-2.241	-2.267	-2.230
	.270	-1.286	-1.620	-1.863	-1.796	-1.834
	.400	-1.114	-1.337	-1.611	-1.637	-1.746
	.620	-1.214	-1.772	-2.049	-2.089	-2.400
	.685	-2.765	-6.013	-6.890	-7.253	-11.000
	.693	-2.294	-6.349	-8.269	-8.826	-11.279
	.700	-1.399	-4.518	-6.300	-6.158	-9.145
	.720	-.670	-1.923	-2.732	-3.222	-4.521
	.750	-.623	-1.087	-1.684	-2.267	-3.423
	.800	-.491	-.547	-.942	-1.592	-2.748
	.900	-.418	-.178	-.332	-.853	-1.903
	.980	-.398	-.059	-.007	-.242	-.764
Lower	.025	.484	.705	.716	.688	.566
	.120	.822	.823	.763	.675	.539
	.220	.763	.757	.743	.700	.566
	.300	.676	.692	.563	.618	.471
	.620	.749	.803	.729	.592	-.116
	.750	.855	.889	.670	.592	.436
	.850	.530	.731	.617	.484	.348
	.950	.378	.533	.484	.337	.184

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TABLE 13 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 30°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = .004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)	Surface	x/c	0.221 (Wing, flap, or aileron)	0.426	0.640	0.800	0.918
α = 18.6°											
.032	-.192	.721	-.303	.160	Upper	.010	-5.326	-3.[illegible]52	-3.733	-3.926	-3.627
.053	-.292	.521	-.415	-.040		.080	-1.916	-2.[illegible]90	-3.552	-3.767	-2.341
.100	-.199	.314	-.540	-.287		.130	-3.342	-3.[illegible]98	-3.332	-3.438	-3.753
.145	-.172	.227	-.461	-.327		.145	-10.243	-9.[illegible]41	-8.647	-9.115	-10.027
.189	-.093	.254	-.435	-.274		.155	-4.433	-4.[illegible]34	-4.788	-4.808	-4.682
.234	-.093	.280	-.119	-.314		.180	-2.987	-3.[illegible]25	-3.472	-3.411	-3.641
.280	-.153	.287	.138	-.347		.220	-1.848	-2.[illegible]70	-2.551	-2.608	-2.626
.326	-.153	.314	.079	-.421		.270	-1.446	-1.[illegible]23	-2.090	-2.114	-2.175
.371	-.351	.414	-.250	-.487		.400	-1.077	-1.[illegible]02	-1.709	-1.791	-2.029
.392	-.420	.455	-.869	-.621		.620	-1.084	-1.[illegible]49	-2.017	-2.206	-2.666
.413	-.497	.521	-1.133	.174		.685	-2.789	-4.[illegible]41	-5.515	-6.401	-11.857
.434	-.557	.574	-1.765	.628		.693	-2.605	-5.[illegible]88	-6.851	-7.765	-12.215
.457	-.544	.570	-1.396	.701		.700	-1.603	-3.[illegible]19	-5.242	-5.552	-10.020
.480	-.524	.560	-1.120	.661		.720	-.750	-1.496	-2.270	-3.023	-5.000
.502	-.544	.550	-1.041	.661		.750	-.559	-.801	-1.382	-2.127	-3.740
.551	-.438	.540	-.975	.621		.800	-.518	-.401	-.741	-1.475	-2.891
.585	-.405	.534	-1.093	.701		.900	-.355	-.194	-.234	-.659	-1.784
.592	-.358	.494	-1.139	-1.115		.980	-.307	-.047	.093	-.237	-.584
.613	-.252	.394	-.810	-.073	Lower	.025	.709	.[illegible]21	.808	.777	.597
.634	-.219	.280	-.514	-.307		.120	.900	.[illegible]35	.768	.724	.451
.655	-.199	.154	-.316	-.321		.220	.859	.[illegible]95	.781	.724	.524
.675	-.119	.027	-.178	-.247		.300	.736	.[illegible]41	.714	.672	.451
.696	-.066	.053	-.059	-.087		.620	.791	.[illegible]15	.735	.626	-.172
.774	.020	.214	.072	-.045		.750	.873	.[illegible]61	.681	.626	.385
.852	-.086	.100	-.053	.000		.850	.675	.[illegible]35	.648	.527	.345
.930	-.007	.140	-.007	.140		.950	.457	.[illegible]34	.481	.329	.219
α = 22.9°											
.032	-.239	.820	-.481	.047	Upper	.010	-9.095	-4.[illegible]41	-4.539	-4.220	-4.262
.053	-.321	.666	-.588	-.128		.080	-2.137	-3.413	-4.700	-4.213	-2.912
.100	-.198	.410	-.654	-.397		.130	-3.459	-3.012	-3.557	-3.606	-3.151
.145	-.177	.316	-.601	-.424		.145	-9.983	-8.244	-7.565	-6.784	-8.333
.189	-.075	.329	-.568	-.370		.155	-4.253	-4.431	-4.505	-3.900	-3.867
.234	-.075	.363	-.180	-.424		.180	-2.811	-3.080	-3.382	-2.878	-3.048
.280	-.143	.383	.120	-.477		.220	-1.769	-2.239	-2.488	-2.150	-2.155
.326	-.211	.390	.053	-.572		.270	-1.502	-1.[illegible]82	-2.004	-1.749	-1.739
.371	-.436	.504	-.474	-.666		.400	-1.102	-1.[illegible]25	-1.540	-1.382	-1.848
.392	-.525	.565	-1.068	-.874		.620	-1.028	-1.[illegible]39	-1.378	-1.202	-1.596
.413	-.607	.619	-1.349	.182		.685	-3.659	-3.[illegible]18	-1.190	-2.330	-4.283
.434	-.641	.639	-2.097	.666		.693	-3.579	-4.[illegible]28	-2.374	-2.718	-4.194
.457	-.580	.610	-1.589	.753		.700	-2.317	-2.[illegible]91	-1.842	-1.823	-3.601
.480	-.518	.590	-1.342	.719		.720	-1.162	-1.[illegible]57	-1.009	-1.128	-2.066
.502	-.505	.570	-1.209	.699		.750	-.755	-.[illegible]87	-.955	-1.082	-1.753
.551	-.348	.565	-1.088	.666		.800	-.521	-.[illegible]45	-.908	-1.015	-1.548
.585	-.307	.551	-1.269	.679		.900	-.401	-.[illegible]30	-.894	-.888	-1.268
.592	-.266	.545	-1.496	-1.486		.980	-.367	-.[illegible]95	-.746	-.855	-1.023
.613	-.164	.437	-1.102	-.550	Lower	.025	.795	.[illegible]67	.820	.808	.648
.634	-.150	.289	-.688	-.424		.120	.895	.[illegible]88	.773	.755	.552
.655	-.130	.168	-.387	-.336		.220	.868	.[illegible]54	.827	.775	.627
.675	-.082	.047	-.187	-.134		.300	.801	.[illegible]87	.760	.721	.559
.696	-.034	.094	-.080	.027		.620	.801	.[illegible]34	.740	.628	.184
.774	-.020	.222	.067	.168		.750	.861	.[illegible]08	.659	.608	.532
.852	-.082	.134	-.053	.061		.850	.708	.[illegible]46	.585	.454	.368
.930	-.007	.155	.000	.141		.950	.474	.[illegible]04	.296	.100	.095

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TABLE 13 Continued
(c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 37°$; $\delta_{a,R} = 30°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = -1.7°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.267	.299	.267	.282
.053	.032	.071	.051	.064
.100	-.102	-.045	-.115	-.109
.145	-.108	-.065	-.064	-.071
.189	-.006	-.032	-.006	.000
.234	-.057	.052	-.013	.006
.280	-.064	.078	-.025	.019
.326	-.032	.117	-.025	.051
.371	-.121	.130	-.096	.128
.392	-.019	.155	-.135	.346
.413	-.185	.188	-.178	-.141
.434	-.223	.182	-.446	-.231
.457	-.287	.210	-.554	-.090
.480	-.350	.240	-.509	.058
.502	-.427	.270	-.554	.218
.551	-.458	.300	-.745	.378
.585	-.439	.338	-.923	.359
.592	-.408	.325	-1.083	-1.045
.613	-.325	.260	-.917	-.750
.634	-.287	.169	-.669	-.814
.655	-.223	.130	-.452	-.179
.675	-.127	.052	-.312	-.096
.696	-.064	.032	-.229	-.045
.774	-.045	.078	-.089	.019
.852	-.032	.019	-.006	-.128
.930	.070	-.208	.096	-.295

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.935	.858	.827	.841	.821
	.080	.389	.292	.244	.242	.299
	.130	-.468	-.715	-.833	-.751	-.739
	.145	-4.821	-4.600	-4.429	-4.865	-4.349
	.155	-1.831	-1.878	-1.923	-1.796	-1.445
	.180	-1.383	-1.247	-1.385	-1.216	-1.286
	.220	-.850	-.962	-1.032	-1.057	-.853
	.270	-.711	-.871	-.942	-.790	-.675
	.400	-.757	-.897	-1.000	-.872	-.688
	.620	-1.363	-1.546	-1.500	-1.191	-.930
	.685	-7.100	-6.913	-5.949	-3.400	-3.598
	.693	-7.047	-7.497	-7.513	-5.247	-3.814
	.700	-4.544	-5.425	-5.705	-3.655	-3.076
	.720	-2.233	-2.404	-2.359	-1.770	-1.248
	.750	-1.554	-1.481	-1.417	-1.261	-.904
	.800	-1.041	-.877	-.808	-.815	-.777
	.900	-.645	-.273	-.404	-.458	-.656
	.980	-.033	.201	-.109	-.102	-.478
Lower	.025	-.356	-.143	.090	.178	-.038
	.120	-.310	-.104	.064	.115	-.102
	.220	-.198	-.117	.038	.076	-.038
	.300	.033	-.149	-.038	.013	-.070
	.620	.619	.546	.141	-.166	-.197
	.750	.883	.754	.173		.025
	.850	.724	.754	.462	.166	.045
	.950	.560	.591	.404	.217	-.057

$\alpha = 5.5°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.086	.464	.145	.318
.053	-.138	.239	-.063	.065
.100	-.211	.066	-.196	-.117
.145	-.165	.007	-.164	-.097
.189	-.079	.060	-.114	-.045
.234	-.112	.093	-.013	-.052
.280	-.112	.133	.063	-.052
.326	-.099	.133	.013	-.032
.371	-.191	.199	-.152	-.013
.392	-.250	.250	-.285	.149
.413	-.316	.292	-.563	.201
.434	-.389	.332	-.848	.247
.457	-.408	.350	-.797	.383
.480	-.468	.370	-.709	.455
.502	-.560	.390	-.683	.474
.551	-.514	.410	-.759	.500
.585	-.494	.438	-.822	.546
.592	-.474	.431	-.797	-.812
.613	-.356	.338	-.652	-.650
.634	-.310	.245	-.481	-.546
.655	-.257	.119	-.348	-.461
.675	-.158	-.007	-.215	-.169
.696	-.092	.027	-.133	.000
.774	-.033	.093	.000	.045
.852	-.046	.053	-.025	-.130
.930	.046	-.119	.038	-.123

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.586	.504	.474	.519	.560
	.080	-.198	-.325	-.383	-.361	-.263
	.130	-1.409	-1.711	-1.852	-1.695	-1.739
	.145	-6.909	-6.837	-6.451	-6.864	-6.546
	.155	-2.845	-3.037	-3.073	-2.904	-2.595
	.180	-2.101	-1.989	-2.170	-1.936	-2.068
	.220	-1.324	-1.466	-1.579	-1.581	-1.469
	.270	-1.014	-1.267	-1.377	-1.227	-1.199
	.400	-.883	-1.141	-1.319	-1.202	-1.172
	.620	-1.133	-1.658	-1.754	-1.518	-1.495
	.685	-3.991	-6.790	-6.425	-3.966	-5.447
	.693	-3.543	-7.440	-8.069	-6.105	-6.290
	.700	-1.996	-5.298	-6.107	-4.378	-5.249
	.720	-.902	-2.301	-2.592	-2.271	-2.457
	.750	-.790	-1.379	-1.605	-1.651	-1.837
	.800	-.685	-.782	-.936	-1.151	-1.488
	.900	-.560	-.252	-.468	-.645	-1.146
	.980	-.454	.119	-.065	-.209	-.606
Lower	.025	.158	.398	.364	.361	.112
	.120	.224	.345	.299	.215	.007
	.220	.547	.444	.429	.291	.145
	.300	.645	.570	.578	.519	.421
	.620	.718	.716	.617	.493	.086
	.750	.836	.802	.533	.462	.415
	.850	.626	.676	.565	.335	.290
	.950	.395	.544	.416	.202	.079

$\alpha = 13.0°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.060	.569	-.080	.267
.053	-.227	.347	-.252	.080
.100	-.180	.170	-.378	-.160
.145	-.140	.085	-.332	-.187
.189	-.087	.118	-.279	-.147
.234	-.114	.170	-.080	-.160
.280	-.120	.209	.106	-.174
.326	-.134	.216	.040	-.247
.371	-.287	.294	-.199	-.260
.392	-.340	.345	-.637	-.234
.413	-.387	.438	-.975	.214
.434	-.447	.432	-1.293	.528
.457	-.467	.440	-1.101	.634
.480	-.481	.450	-.955	.594
.502	-.541	.460	-.889	.561
.551	-.501	.470	-.928	.588
.585	-.434	.484	-.928	.654
.592	-.421	.412	-.935	-.868
.613	-.307	.347	-.716	-.425
.634	-.274	.229	-.477	-.347
.655	-.247	.118	-.325	-.361
.675	-.140	-.020	-.192	-.247
.696	-.093	.020	-.099	-.080
.774	.000	.164	.060	.093
.852	-.060	.078	-.013	-.040
.930	.013	.026	.027	.047

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.265	-1.439	-1.803	-1.863	-1.128
	.080	-1.061	-1.066	-1.249	-1.366	-1.149
	.130	-2.553	-2.767	-3.165	-3.130	-3.085
	.145	-8.820	-8.687	-8.694	-9.595	-8.954
	.155	-3.773	-4.049	-4.380	-4.463	-3.799
	.180	-2.553	-2.643	-3.032	-2.997	-2.918
	.220	-1.664	-1.917	-2.210	-2.301	-2.030
	.270	-1.260	-1.563	-1.810	-1.790	-1.636
	.400	-1.061	-1.276	-1.576	-1.592	-1.556
	.620	-1.147	-1.688	-1.950	-1.923	-1.990
	.685	-2.646	-5.776	-6.410	-4.748	-6.911
	.693	-2.202	-6.325	-8.106	-7.003	-8.033
	.700	-1.320	-4.494	-6.177	-5.146	-6.838
	.720	-.637	-1.884	-2.651	-2.765	-3.392
	.750	-.584	-1.073	-1.629	-2.036	-2.631
	.800	-.471	-.556	-.955	-1.459	-2.190
	.900	-.405	-.196	-.447	-.836	-1.636
	.980	-.371	-.020	-.067	-.325	-.741
Lower	.025	.464	.680	.708	.690	.528
	.120	.822	.798	.748	.696	.541
	.220	.782	.739	.741	.703	.568
	.300	.670	.680	.668	.597	.427
	.620	.763	.778	.681	.517	-.120
	.750	.849	.863	.588	.484	.341
	.850	.643	.713	.561	.371	.260
	.950	.398	.530	.447	.239	.067

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TABLE 13 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 37°; $\delta_{a,R}$ = 30°; h_s/c = 0.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 18.7°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.195	.721	-.325	.164
.053	-.309	.514	-.460	-.034
.100	-.202	.314	-.528	-.274
.145	-.168	.200	-.501	-.314
.189	-.094	.240	-.470	-.245
.234	-.094	.280	-.149	-.371
.280	-.134	.300	.122	-.348
.326	-.168	.307	.068	-.443
.371	-.363	.421	-.305	-.484
.392	.108	.470	-.907	-.607
.413	-.498	.534	-1.165	.198
.434	-.558	.548	-1.815	.634
.457	-.551	.545	-1.388	.736
.480	-.531	.540	-1.131	.696
.502	-.551	.535	-1.043	.661
.551	-.437	.530	-.989	.634
.585	-.397	.528	-1.117	.689
.592	-.383	.521	-1.151	-1.118
.613	-.269	.414	-.840	-.425
.634	-.229	.287	-.569	-.321
.655	-.202	.167	-.352	-.321
.675	-.101	.060	-.176	-.275
.696	-.054	.067	-.061	-.075
.774	.007	.214	.068	-.035
.852	-.067	.093	-.041	.007
.930	-.020	.120	-.007	.136

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-5.215	-3.192	-3.785	-3.731	-3.671
	.080	-1.856	-2.250	-3.567	-3.562	-2.320
	.130	-3.265	-3.078	-3.389	-3.501	-3.779
	.145	-9.963	[illegible]	-8.749	-9.521	-10.113
	.155	-4.294	[illegible]	-4.821	-4.923	-4.673
	.180	-2.898	[illegible]	-3.498	-3.487	-3.644
	.220	-1.816	[illegible]	-2.544	-2.654	-2.616
	.270	-1.376	[illegible]	-2.094	-2.113	-2.138
	.400	-1.035	[illegible]	-1.691	-1.788	-1.977
	.620	-1.035	[illegible]	-1.957	-2.092	-2.448
	.685	-2.738	[illegible]	-5.258	-4.381	-8.042
	.693	-2.544	[illegible]	-6.744	-6.358	-9.501
	.700	-1.556	[illegible]	-5.135	-4.713	-8.224
	.720	-.721	[illegible]	-2.203	-2.695	-4.229
	.750	-.548	[illegible]	-1.357	-2.031	-3.295
	.800	-.494	[illegible]	-.757	-1.436	-2.636
	.900	-.361	[illegible]	-.245	-.745	-1.755
	.980	-.321	-.047	.048	-.271	-.686
Lower	.025	.688	.878	.798	.792	.592
	.120	.855	.878	.764	.684	.451
	.220	.835	[illegible]	.805	.745	.524
	.300	.728	[illegible]	.716	.650	.457
	.620	.768	[illegible]	.723	.528	-.134
	.750	.855	[illegible]	.621	.521	.329
	.850	.661	[illegible]	.634	.440	.269
	.950	.461	[illegible]	.505	.257	.121

α = 22.8°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.227	.809	-.451	
.053	-.347	.595	-.549	-.159
.100	-.220	.387	-.621	-.358
.145	-.167	.297	-.569	-.438
.189	-.073	.318	-.510	-.345
.234	-.053	.339	-.144	-.438
.280	-.107	.367	.137	-.451
.326	-.207	.401	.072	-.584
.371	-.434	.498	-.321	-.676
.392	-.510	.550	-1.020	-.895
.413	-.588	.602	-1.302	.146
.434	-.641	.629	-2.015	.637
.457	-.561	.605	-1.576	.710
.480	-.494	.595	-1.367	.690
.502	-.487	.585	-1.197	.650
.551	-.321	.575	-1.073	.623
.585	-.254	.574	-1.197	.643
.592	-.227	.540	-1.511	-1.479
.613	-.147	.408	-1.092	-.480
.634	-.127	.304	-.641	-.371
.655	-.120	.180	-.347	-.272
.675	-.047	.028	-.177	-.066
.696	-.013	.069	-.052	.053
.774	.013	.090	.065	.159
.852	-.067	.118	-.039	.066
.930	[illegible]	.145	-.007	.139

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-9.089	[illegible]	-4.615	-4.454	-4.481
	.080	-2.061	[illegible]	-4.814	-4.474	-3.699
	.130	-3.313	-3.119	-3.554	-3.905	-3.579
	.145	-9.464	-8.509	-7.540	-7.352	-9.689
	.155	-3.991	[illegible]	-4.536	-4.291	-4.647
	.180	-2.687	-3.265	-3.422	-3.199	-3.686
	.220	-1.778	-2.380	-2.546	-2.440	-2.698
	.270	-1.528	-1.910	-2.095	-2.008	-2.230
	.400	-1.067	-1.439	-1.631	-1.616	-2.023
	.620	-1.054	-1.646	-1.618	-1.544	-2.197
	.685	-3.675	-3.908	-2.440	-2.387	-6.210
	.693	-3.596	-4.213	-3.753	-2.878	-7.038
	.700	-2.305	-3.064	-2.812	-2.159	-5.923
	.720	-1.133	-1.287	-1.220	-1.302	-2.865
	.750	-.744	-.761	-.928	-1.079	-2.190
	.800	-.487	-.435	-.749	-.942	-1.729
	.900	-.329	-.263	-.610	-.739	-1.269
	.980	-.316	-.111	-.365	-.641	-.694
Lower	.025	.797	.872	.796	.785	.601
	.120	.902	.844	.729	.706	.474
	.220	.856	[illegible]	.782	.739	.574
	.300	.803	[illegible]	.716	.687	.487
	.620	.803	.837	.663	.563	.040
	.750	.869	[illegible]	.577	.517	.387
	.850	.711	[illegible]	.550	.392	.294
	.950	.501	[illegible]	.358	.098	.067

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TABLE 13 Continued

(d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 18°$; $\delta_{a,R} = 30°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = -1.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.244	.323	.267	.316
.053	.032	.099	.070	.053
.100	-.115	-.040	-.108	-.086
.145	-.096	-.086	-.070	-.040
.189	-.038	-.020	-.019	.007
.234	-.077	.040	-.006	.013
.280	-.083	.072	-.025	.033
.326	-.058	.092	-.032	.079
.371	-.128	.138	-.025	.125
.392	-.154	.142	-.006	.329
.413	-.179	.145	-.178	-.244
.434	-.231	.191	-.427	-.303
.457	-.269	.220	-.541	-.112
.480	-.321	.250	-.509	.026
.502	-.417	.280	-.548	.250
.551	-.455	.310	-.745	.408
.585	-.449	.323	-.904	.389
.592	-.429	.323	-1.076	-1.080
.613	-.327	.277	-.898	-.790
.634	-.353	.211	-.637	-.777
.655	-.295	.132	-.458	-.171
.675	-.218	.053	-.299	-.079
.696	-.128	.013	-.197	-.040
.774	-.077	.079	-.089	.007
.852	-.064	-.020	.000	-.112
.930	-.032	-.217	.096	-.323

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.936	.869	.856	.834	.827
	.080	.429	.316	.277	.248	.314
	.130	-.397	-.698	-.803	-.732	-.692
	.145	-4.545	-4.498	-4.373	-4.674	-4.205
	.155	-1.692	-1.851	-1.851	-1.719	-1.404
	.180	-1.282	-1.205	-1.344	-1.165	-1.218
	.220	-.788	-.922	-.988	-1.012	-.808
	.270	-.635	-.850	-.896	-.751	-.628
	.400	-.692	-.883	-.962	-.802	-.603
	.620	-1.263	-1.521	-1.436	-.987	-.686
	.685	-6.808	-6.849	-5.657		
	.693	-6.750	-7.416	-7.389	-2.528	-2.256
	.700	-4.282	-5.420	-5.591	-2.324	-2.154
	.720	-2.128	-2.410	-2.292	-1.267	-.859
	.750	-1.481	-1.488	-1.429	-1.006	-.673
	.800	-.994	-.889	-.915	-.707	-.571
	.900	-.590	-.303	-.698	-.439	-.487
	.980	.006	.158	-.296	-.140	-.346
Lower	.025	-.397	-.250	.040	.096	-.090
	.120	-.314	-.198	.020	.032	-.160
	.220	-.231	-.244	-.013	.000	-.115
	.300	.045	-.323	-.059	-.070	-.135
	.620	.654	.698	.158	-.287	-.269
	.750	.827	.850	.033	-.121	-.141
	.850	.673	.764	.454	.051	-.128
	.950	.538	.580	.356	.102	-.167

$\alpha = 5.6°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.072	.491	.113	.321
.053	-.125	.245	-.086	.115
.100	-.184	.073	-.232	-.115
.145	-.138		-.186	-.077
.189	-.086	.053	-.126	-.038
.234	-.092	.113	-.060	-.038
.280	-.099	.119	.027	-.032
.326	-.112	.139	.007	-.026
.371	-.204	.225	-.153	-.006
.392	-.255	.265	-.298	.147
.413	-.303	.305	-.564	.154
.434	-.369	.345	-.882	.205
.457	-.389	.365	-.829	.359
.480	-.448	.385	-.749	.449
.502	-.527	.405	-.729	.455
.551	-.514	.425	-.816	.487
.585	-.481	.444	-.882	.526
.592	-.441	.424	-.882	-.846
.613	-.362	.332	-.703	-.600
.634	-.303	.232	-.537	-.538
.655	-.270	.133	-.365	-.442
.675	-.171	.033	-.239	-.135
.696	-.105	.033	-.146	.000
.774	-.033	.093	.000	.013
.852	-.046	.053	-.033	-.141
.930	.033	-.113	.046	-.167

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.601	.564	.526	.590	.639
	.080	-.187	-.279	-.308	-.351	-.171
	.130	-1.362	-1.651	-1.718	-1.698	-1.600
	.145	-6.918	-6.711	-6.179	-6.996	-6.178
	.155	-2.845	-2.984	-2.897	-2.905	-2.397
	.180	-2.063	-1.950	-2.026	-1.936	-1.936
	.220	-1.302	-1.432	-1.474	-1.598	-1.317
	.270	-1.015	-1.233	-1.276	-1.220	-1.087
	.400	-.901	-1.094	-1.205	-1.154	-1.014
	.620	-1.162	-1.585	-1.583	-1.293	-1.106
	.685	-4.107	-6.757	-6.026		
	.693	-3.646	-7.387	-7.628	-3.070	-3.201
	.700	-2.050	-5.252	-5.795	-2.871	-3.280
	.720	-.935	-2.281	-2.468	-1.684	-1.535
	.750	-.828	-1.379	-1.571	-1.346	-1.225
	.800	-.714	-.809	-1.032	-1.001	-1.014
	.900	-.614	-.259	-.712	-.637	-.771
	.980	-.481	.119	-.263	-.318	-.402
Lower	.025	-.147	.391	.340	.298	.013
	.120	.341	.318	.256	.080	-.092
	.220	.641	.365	.308	.199	.086
	.300	.601	.524	.526	.477	.290
	.620	.701	.710	.551	.312	.053
	.750	.841	.816	.385	.192	.263
	.850	.628	.703	.500	-.033	.119
	.950	.381	.557	.333	-.093	-.053

$\alpha = 13.1°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.087	.619	-.103	.271
.053	-.247	.408	-.302	.047
.100	-.200	.178	-.412	-.149
.145	-.147	.092	-.364	-.149
.189	-.080	.125	-.295	-.108
.234	-.087	.198	-.110	-.149
.280	-.134	.211	.062	-.169
.326	-.154	.217	.014	-.223
.371	-.287	.316	-.220	-.244
.392	-.340	.360	-.666	-.190
.413	-.381	.415	-1.016	.190
.434	-.447	.468	-1.360	.528
.457	-.454	.470	-1.154	.616
.480	-.481	.475	-.996	.582
.502	-.548	.480	-.913	.535
.551	-.487	.481	-.975	.582
.585	-.441	.421	-.927	.643
.592	-.421	.468	-.920	-.874
.613	-.321	.362	-.728	-.420
.634	-.280	.250	-.536	-.366
.655	-.247	.138	-.357	-.393
.675	-.147	.026	-.206	-.237
.696	-.093	.046	-.110	-.061
.774	-.013	.151	.027	.115
.852	-.080	.092	-.041	-.047
.930	.000	.033	-.007	.020

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.252	-1.304	-1.571	-1.628	-.454
	.080	-1.054	-1.047	-1.205	-1.360	-1.048
	.130	-2.513	-2.707	-3.061	-3.098	-2.938
	.145	-8.767	-8.634	-8.539	-9.581	-8.640
	.155	-3.733	-4.011	-4.259	-4.368	-3.653
	.180	-2.553	-2.608	-2.946	-2.926	-2.784
	.220	-1.651	-1.877	-2.113	-2.253	-1.930
	.270	-1.253	-1.535	-1.740	-1.751	-1.542
	.400	-1.068	-1.238	-1.476	-1.504	-1.389
	.620	-1.127	-1.581	-1.767	-1.600	-1.549
	.685	-2.858	-6.151	-6.297		
	.693	-2.241	-6.461	-7.902	-3.516	-4.100
	.700	-1.373	-4.511	-6.020	-3.386	-4.287
	.720	-.676	-1.923	-2.614	-2.033	-2.230
	.750	-.617	-1.093	-1.700	-1.635	-1.816
	.800	-.491	-.580	-1.097	-1.250	-1.536
	.900	-.431	-.178	-.738	-.810	-1.149
	.980	-.405	-.020	-.230	-.453	-.561
Lower	.025	.458	.678	.677	.646	.481
	.120	.807	.771	.752	.673	.548
	.220	.776	.731	.752	.652	.548
	.300	.663	.665	.664	.556	.414
	.620	.743	.777	.623	.337	-.140
	.750	.829	.850	.311	.179	.207
	.850	.617	.705	.508	-.041	.073
	.950	.405	.527	.406	-.103	-.067

TABLE 13 Concluded
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 18°$; $\delta_{a,R} = 30°$; $h_s/c = 0.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = 18.7°$											
.032	-.177	.742	-.270	.173	Upper	.010	-5.173	-3.166	-3.618	-3.411	-3.301
.053	-.334	.529	-.402			.080	-1.876	-2.205	-3.334	-3.155	-1.773
.100	-.198	.302	-.474	-.284		.130	-3.311	-3.125	-3.383	-3.313	-3.792
.145	-.170	.206	-.461	-.304		.145	-10.096	-9.382	-8.737	-9.200	-10.059
.189	-.102	.234	-.402	-.256		.155	-4.340	-4.669	-4.794	-4.683	-4.562
.234	-.095	.288	-.125	-.318		.180	-2.925	-3.125	-3.438	-3.260	-3.505
.280	-.143	.302	.119	-.353		.220	-1.815	-2.266	-2.511	-2.450	-2.482
.326	-.191	.330	.026	-.429		.270	-1.361	-1.806	-2.041	-1.936	-1.998
.371	-.361	.433	-.263	-.491		.400	-1.043	-1.380	-1.639	-1.581	-1.780
.392	-.420	.520	-.843	-.609		.620	-1.043	-1.676	-1.875	-1.686	-1.950
.413	-.491	.549	-1.113	.173		.685	-2.898	-4.993	-5.147		
.434	-.566	.584	-1.732	.616		.693	-2.661	-5.302	-6.641	-3.319	-4.849
.457	-.546	.575	-1.337	.706		.700	-1.645	-3.736	-5.084	-3.273	-5.087
.480	-.511	.568	-1.060	.671		.720	-.731	-1.545	-2.214	-2.055	-2.857
.502	-.539	.560	-.975	.636		.750	-.535	-.845	-1.432	-1.673	-2.394
.551	-.443	.553	-.909	.609		.800	-.508	-.398	-.885	-1.344	-2.080
.585	-.389	.549	-1.041	.671		.900	-.359	-.165	-.519	-.823	-1.541
.592	-.368	.522	-1.080	-1.162		.980	-.366	-.069	-.118	-.415	-.716
.613	-.259	.419	-.790	-.425	Lower	.025	.704	.824	.796	.757	.586
.634	-.225	.316	-.533	-.346		.120	.880	.817	.726	.672	.484
.655	-.205	.179	-.323	-.325		.220	.819	.797	.768	.711	.525
.675	-.116	.041	-.171	-.228		.300	.745	.749	.692	.632	.430
.696	-.075	.076	-.053	-.062		.620	.779	.824	.629	.395	-.123
.774	-.007	.220	.079	.104		.750	.874	.865	.436	.257	.198
.852	-.082	.103	-.040	-.007		.850	.670	.769	.526	.119	.095
.930	-.007	.130	-.020	.125		.950	.454	.556	.401	.000	-.082
$\alpha = 22.8°$											
.032	-.225	.816	-.481	.054	Upper	.010	-9.416	-4.275	-4.519	-4.657	-4.284
.053	-.345	.650	-.591	-.114		.080	-2.129	-4.095	-4.687	-4.670	-3.137
.100	-.199	.394	-.646	-.383		.130	-3.407	-3.175	-3.490	-4.038	-3.515
.145	-.172	.318	-.604	-.444		.145	-9.746	-8.509	-7.558	-7.933	-9.430
.189	-.073	.304	-.536	-.370		.155	-4.135	-4.628	-4.512	-4.595	-4.456
.234	-.053	.360	-.165	-.437		.180	-2.768	-3.251	-3.355	-3.379	-3.495
.280	-.106	.360	.137	-.464		.220	-1.868	-2.373	-2.474	-2.582	-2.480
.326	-.186	.394	.055	-.578		.270	-1.580	-1.902	-2.010	-2.102	-2.009
.371	-.411	.477	-.364	-.666		.400	-1.085	-1.439	-1.533	-1.683	-1.777
.392	-.500	.550	-1.058	-.881		.620	-1.085	-1.639	-1.486	-1.648	-1.916
.413	-.584	.623	-1.374	.168		.685	-3.880	-4.033	-1.755		
.434	-.623	.636	-2.095	.659		.693	-3.668	-4.157	-2.965	-2.287	-4.642
.457	-.557	.610	-1.635	.726		.700	-2.376	-2.961	-2.253	-2.205	-4.847
.480	-.477	.590	-1.429	.693		.720	-1.195	-1.238	-1.029	-1.497	-2.633
.502	-.464	.580	-1.250	.652		.750	-.797	-.706	-.814	-1.271	-2.155
.551	-.298	.570	-1.113	.625		.800	-.536	-.408	-.679	-1.078	-1.804
.585	-.245	.567	-1.250	.652		.900	-.343	-.208	-.598	-.762	-1.273
.592	-.225	.553	-1.580	-1.486		.980	-.350	-.090	-.464	-.646	-.610
.613	-.159	.450	-1.161	-.425	Lower	.025	.797	.872	.780	.769	.597
.634	-.133	.318	-.680	-.397		.120	.920	.851	.740	.687	.451
.655	-.099	.187	-.357	-.269		.220	.879	.823	.773	.714	.537
.675	-.040	.062	-.172	-.087		.300	.797	.782	.693	.646	.464
.696	-.013	.090	-.062	.047		.620	.810	.851	.639	.426	-.013
.774	[illegible]	.228	.048	.148		.750	.893	.913	.417	.288	.279
.852	-.066	.125	-.055	.047		.850	.721	.782	.524	.172	.179
.930	.000	.159	-.007	.128		.950	.529	.560	.336	-.069	.007

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TABLE 14 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.7$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
$\alpha = -1.4°$				
.032	.270	.287	.298	.304
.053	.038	.054	.049	.056
.100	-.094	-.048	-.091	-.105
.145	-.094	-.096	-.073	-.043
.189	-.019	-.036	[illegible]	-.025
.234	-.057	.030	-.030	-.006
.280	-.050	.060	-.067	.025
.326	-.050	.072	-.067	.062
.371	-.094	.102	-.103	.143
.392	-.082	-.072	.037	.347
.413	-.126	.119	-.067	-.087
.434	-.170	.119	-.268	-.261
.457	-.195	.200	-.341	-.211
.480	-.233	.200	-.274	-.112
.502	-.321	.200	-.256	.037
.551	-.358	.200	-.213	.236
.585	-.383	.203	-.286	.199
.592	-.383	.215	-.383	-.372
.613	-.346	.143	-.353	-.056
.634	-.333	.072	-.329	-.360
.655	-.308	-.030	-.329	-.490
.675	-.239	-.143	-.286	-.527
.696	-.195	-.131	-.213	-.465
.774	.025	.018	.024	-.031
.852	-.050	-.060	.012	-.112
.930	.006	-.167	.024	-.136
$\alpha = 5.8°$				
.032	.106	.455	.162	.337
.053	-.087	.228	-.075	.083
.100	-.194	.038	-.206	-.096
.145	-.137	-.025	-.162	-.064
.189	-.050	.038	-.100	-.025
.234	-.087	.089	-.037	-.038
.280	-.094	.120	.012	-.032
.326	-.087	.120	-.012	.000
.371	-.162	.196	-.106	.045
.392	-.200	.230	-.206	.197
.413	-.244	.272	-.450	.159
.434	-.318	.297	-.718	.178
.457	-.318	.300	-.637	.280
.480	-.350	.320	-.531	.382
.502	-.437	.340	-.450	.433
.551	-.418	.360	-.325	.458
.585	-.431	.373	-.362	.478
.592	-.425	.373	-.462	-.427
.613	-.381	.278	-.406	-.229
.634	-.362	.152	-.400	-.420
.655	-.350	.019	-.425	-.592
.675	-.275	-.152	-.381	-.592
.696	-.225	-.183	-.287	-.478
.774	-.130	.051	.000	-.076
.852	-.050	.025	-.012	-.083
.930	.037	-.101	.006	-.115
$\alpha = 13.2°$				
.032	-.092	.607	-.091	.237
.053	-.288	.392	-.260	.013
.100	-.222	.202	-.364	-.205
.145	-.170	.108	-.325	-.205
.189	-.105	.152	-.273	-.167
.234	-.118	.196	-.117	-.179
.280	-.124	.202	.052	-.192
.326	-.150	.209	.019	-.218
.371	-.288	.316	-.214	-.224
.392	-.340	.360	-.572	-.167
.413	-.386	.405	-.884	.192
.434	-.432	.462	-1.189	.513
.457	-.419	.470	-.942	.590
.480	-.445	.475	-.747	.551
.502	-.536	.470	-.624	.506
.551	-.471	.460	-.455	.532
.585	-.445	.443	-.455	.609
.592	-.458	.411	-.520	-.468
.613	-.386	.304	-.481	-.410
.634	-.353	.164	-.468	-.468
.655	-.353	.051	-.474	-.628
.675	-.255	-.114	-.435	-.532
.696	-.216	-.114	-.344	-.391
.774	.026	.101	-.026	-.038
.852	-.039	.057	-.052	-.115
.930	.000	.006	-.032	.019

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
$\alpha = -1.4°$						
Upper	.010	.968	.854	.844	.840	.798
	.080	.524	.406	.354	.323	.333
	.130	-.271	-.448	-.602	-.584	-.610
	.145	-4.124	-3.727	-3.722	-4.284	-4.022
	.155	-1.461	-1.409	-1.495	-1.479	-1.207
	.180	-1.066	-.872	-1.055	-.986	-1.138
	.220	-.586	-.597	-.732	-.852	-.760
	.270	-.431	-.484	-.633	-.609	-.597
	.400	-.327	-.358	-.608	-.688	-.641
	.620	-.092	-.048	-.707	-.974	-.980
	.685	-.943	-.579	-2.599	-6.293	-5.549
	.693	-.894	-.908	-3.859	-6.725	-4.613
	.700	-.727	-.800	-2.711	-4.096	-3.262
	.720	-.653	-.645	-.788	-1.728	-1.150
	.750	-.549	-.758	-.658	-1.083	-.886
	.800	-.419	-.466	-.664	-.633	-.786
	.900	-.401	-.514	-.670	-.304	-.691
	.980	-.345	-.490	-.540	-.055	-.547
Lower	.025	-.240	-.179	.031	.103	-.031
	.120	-.345	-.149	.012	.073	-.057
	.220	-.284	-.155	-.019	.037	-.044
	.300	-.123	-.179	-.068	-.012	-.094
	.620	.450	.179	.037	-.037	-.226
	.750	.789	.394	.112	.073	.050
	.850	.696	.520	.285	.195	.226
	.950	.388	.311	.248	.292	.258
$\alpha = 5.8°$						
Upper	.010	.654	.633	.573	.599	.599
	.080	-.071	-.127	-.229	-.262	-.200
	.130	-1.160	-1.341	-1.560	-1.530	-1.586
	.145	-6.199	-5.820	-5.699	-6.462	-6.106
	.155	-2.468	-2.467	-2.604	-2.647	-2.285
	.180	-1.769	-1.569	-1.808	-1.761	-1.904
	.220	-1.038	-1.075	-1.261	-1.405	-1.324
	.270	-.750	-.873	-1.019	-1.061	-1.074
	.400	-.519	-.557	-.866	-1.043	-1.080
	.620	-.205	-.139	-.796	-1.536	-1.630
	.685	-.929	-.614	-2.382	-8.117	-10.190
	.693	-.923	-.810	-3.540	-8.716	-9.272
	.700	-.724	-.810	-2.401	-5.619	-7.018
	.720	-.635	-.746	-.758	-2.566	-3.134
	.750	-.641	-.620	-.751	-1.636	-2.210
	.800	-.404	-.430	-.694	-1.049	-1.736
	.900	-.513	-.493	-.586	-.512	-1.274
	.980	-.487	-.418	-.599	-.037	-.487
Lower	.025	.051	.316	.357	.387	.194
	.120	.115	.304	.312	.300	.106
	.220	.474	.297	.293	.287	.112
	.300	.571	.430	.478	.481	.462
	.620	.667	.671	.681	.674	.256
	.750	.795	.740	.739	.724	.599
	.850	.551	.582	.599	.656	.562
	.950	.276	.234	.255	.475	.362
$\alpha = 13.2°$						
Upper	.010	-.092	-.999	-1.308	-1.351	-.863
	.080	-.929	-.898	-1.077	-1.208	-1.073
	.130	-2.316	-2.410	-2.756	-2.768	-2.904
	.145	-8.386	-7.800	-7.673	-8.894	-8.458
	.155	-3.480	-3.536	-3.763	-3.950	-3.467
	.180	-2.335	-2.233	-2.545	-2.586	-2.721
	.220	-1.413	-1.537	-1.776	-1.969	-1.871
	.270	-1.001	-1.164	-1.397	-1.481	-1.498
	.400	-.693	-.715	-1.006	-1.286	-1.400
	.620	-.347	-.247	-.814	-1.982	-2.060
	.685	-.942	-.557	-1.365	-8.108	-11.944
	.693	-.916	-.822	-2.359	-8.920	-10.897
	.700	-.720	-.702	-1.628	-5.620	-8.288
	.720	-.563	-.715	-.750	-2.618	-3.741
	.750	-.602	-.576	-.686	-1.644	-2.603
	.800	-.517	-.373	-.615	-1.033	-2.067
	.900	-.549	-.418	-.519	-.520	-1.681
	.980	-.641	-.386	-.551	-.078	-1.066
Lower	.025	.471	.652	.654	.695	.549
	.120	.811	.822	.750	.754	.602
	.220	.798	.765	.744	.747	.628
	.300	.667	.671	.692	.663	.523
	.620	.759	.772	.750	.708	.111
	.750	.824	.873	.769	.715	.582
	.850	.595	.645	.590	.650	.458
	.950	.288	.316	.250	.474	.216

TABLE 14 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 2.7 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

Fuselage

α = 19.0°

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.192	.695	-.318	.130
.053	-.325	.442	-.468	-.078
.100	-.225	.305	-.526	-.286
.145	-.172	.175	-.487	-.312
.189	-.086	.227	-.416	-.266
.234	-.086	.253	-.169	-.299
.280	-.113	.286	.091	-.318
.326	-.139	.312	.013	-.390
.371	-.332	.409	-.292	-.416
.392	-.390	.470	-.819	-.487
.413	-.458	.526	-1.046	.195
.434	-.524	.559	-1.566	.604
.457	-.517	.560	-1.234	.676
.480	-.471	.540	-.910	.637
.502	-.504	.520	-.747	.611
.551	-.405	.500	-.520	.598
.585	-.358	.487	-.559	.656
.592	-.351	.461	-.591	-.520
.613	-.292	.318	-.526	-.409
.634	-.272	.201	-.474	-.481
.655	-.252	.032	-.468	-.565
.675	-.139	-.097	-.416	-.487
.696	-.099	-.091	-.344	-.351
.774	.040	.149	-.210	[illegible]
.852	-.073	.058	-.091	-.123
.930	-.033	.130	-.091	.149

α = 23.1°

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.250	.760	-.467	.039
.053	-.375	.546	-.574	-.136
.100	-.224	.377	-.654	-.351
.145	-.178	.292	-.588	-.396
.189	-.105	.305	-.541	-.338
.234	-.079	.351	-.194	-.396
.280	-.125	.370	.093	-.403
.326	-.204	.377	-.134	-.513
.371	-.389	.481	-.487	-.585
.392	-.470	.530	-.942	-.760
.413	-.573	.585	-1.249	.188
.434	-.606	.611	-2.003	.656
.457	-.560	.620	-1.436	.708
.480	-.494	.620	-1.142	.682
.502	-.487	.580	-.955	.643
.551	-.349	.540	-.748	.598
.585	-.303	.507	-.835	.650
.592	-.290	.487	-.895	-.767
.613	-.204	.351	-.881	-.377
.634	-.204	.221	-.915	-.598
.655	-.178	.039	-.781	-.837
.675	-.099	-.084	-.574	-.461
.696	-.086	-.039	-.394	-.201
.774	-.007	[illegible]	-.250	.110
.852	-.125	.097	-.107	.150
.930	-.105	.156	-.107	.195

Wing, flap, or aileron

α = 19.0°

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-3.655	-2.670	-3.041	-3.112	-2.977
	.080	-1.653	-1.605	-2.508	-2.774	-1.704
	.130	-3.049	-2.839	-3.015	-3.041	-3.448
	.145	-9.444	-8.478	-7.991	-8.810	-9.251
	.155	-4.031	-4.113	-4.210	-4.320	-4.118
	.180	-2.641	-2.657	-2.911	-2.917	-3.190
	.220	-1.587	-1.826	-2.040	-2.176	-2.235
	.270	-1.139	-1.364	-1.572	-1.676	-1.810
	.400	-.731	-.812	-1.059	-1.312	-1.711
	.620	-.362	-.325	-.721	-1.982	-2.036
	.685	-.962	-.578	-.234	-5.685	-10.975
	.693	-.962	-.747	-1.488	-6.140	-9.947
	.700	-.751	-.754	-1.195	-3.755	-7.493
	.720	-.593	-.656	-.676	-1.644	-3.269
	.750	-.573	-.565	-.624	-.988	-2.268
	.800	-.527	-.396	-.591	-.689	-1.784
	.900	-.540	-.435	-.481	-.559	-1.426
	.980	-.593	-.396	-.559	-.383	-.855
Lower	.025	.678	.799	.780	.806	.643
	.120	.869	.832	.780	.728	.544
	.220	.843	.819	.812	.767	.603
	.300	.724	.721	.715	.689	.537
	.620	.771	.799	.780	.702	.126
	.750	.856	.838	.754	.708	.584
	.850	.626	.676	.624	.624	.497
	.950	.356	.357	.260	.390	.265

α = 23.1°

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-7.523	-3.651	-3.801	-3.713	-3.431
	.080	-1.923	-3.352	-3.898	-3.693	-2.042
	.130	-3.157	-2.781	-3.002	-3.018	-2.384
	.145	-9.381	-7.673	-6.503	-6.811	-6.336
	.155	-3.944	-4.061	-3.710	-3.532	-2.608
	.180	-2.553	-2.735	-2.677	-2.504	-2.061
	.220	-1.533	-1.904	-1.839	-1.790	-1.271
	.270	-1.182	-1.429	-1.423	-1.349	-.955
	.400	-.812	-.910	-.981	-1.008	-.889
	.620	-.468	-.546	-.799	-1.015	-.863
	.685	-1.325	-.923	-.286	-1.930	-2.061
	.693	-1.280	-1.052	-1.618	-2.043	-1.719
	.700	-1.040	-1.072	-1.338	-1.449	-1.475
	.720	-.819	-.832	-.728	-.988	-.915
	.750	-.708	-.715	-.708	-.962	-.929
	.800	-.604	-.572	-.630	-.935	-.929
	.900	-.604	-.624	-.565	-.935	-.935
	.980	-.598	-.578	-.526	-.888	-.869
Lower	.025	.786	.832	.858	.821	.692
	.120	.916	.832	.812	.788	.606
	.220	.871	.832	.819	.788	.685
	.300	.786	.754	.760	.728	.612
	.620	.793	.812	.793	.735	.316
	.750	.851	.871	.799	.694	.626
	.850	.663	.663	.630	.574	.501
	.950	.390	.312	.305	.187	.178

L-921

10X

L-927

TABLE 14 Continued

(b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 4.7 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = -1.3°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.264	.288	.270	.294
.053	.024	.060	.042	.049
.100	-.096	-.042	-.096	-.080
.145	-.096	-.108	-.072	-.037
.189	-.006	-.042	-.012	.000
.234	-.048	.036	-.048	.024
.280	-.042	.066	-.078	.037
.326	-.018	.072	-.072	.086
.371	-.066	.120	-.084	.165
.392	-.060	-.060	.072	.361
.413	-.090	.138	-.072	-.110
.434	-.132	.138	-.234	-.286
.457	-.144	-.140	-.282	-.220
.480	-.198	-.150	-.228	-.110
.502	-.264	-.160	-.156	.031
.551	-.313	-.175	-.078	.257
.585	-.361	.192	-.276	.220
.592	-.379	.180	-.469	-.410
.613	-.373	.114	-.493	-.110
.634	-.355	.030	-.481	-.343
.655	-.349	-.072	-.481	-.478
.675	-.282	-.180	-.481	-.557
.696	-.241	-.192	-.385	-.570
.774	.018	-.018	-.066	-.135
.852	-.036	-.072	-.018	-.110
.930	.030	-.144	-.006	-.135

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.955	.877	.827	.817	.793
	.080	.532	.415	.361	.319	.325
	.130	-.175	-.397	-.514	-.517	-.553
	.145	-3.780	-3.510	-3.424	-4.050	-3.720
	.155	-1.270	-1.274	-1.335	-1.382	-1.118
	.180	-.919	-.769	-.931	-.925	-1.034
	.220	-.460	-.493	-.625	-.793	-.685
	.270	-.290	-.385	-.514	-.577	-.535
	.400	-.157	-.156	-.447	-.625	-.565
	.620	.254	.343	-.514	-.950	-.913
	.685	-.895	-.451	-1.911	-6.154	-5.661
	.693	-.943	-.673	-3.136	-6.779	-4.796
	.700	-.768	-.739	-2.119	-4.075	-3.377
	.720	-.689	-.595	-.698	-1.707	-1.142
	.750	-.689	-.715	-.643	-1.040	-.841
	.800	-.417	-.613	-.570	-.601	-.739
	.900	-.357	-.469	-.631	-.276	-.649
	.980	-.357	-.469	-.527	-.012	-.493
Lower	.025	-.254	-.198	.018	.060	-.036
	.120	-.351	-.156	.000	.012	-.060
	.220	-.314	-.156	-.024	.000	-.060
	.300	-.175	-.204	-.086	-.036	-.090
	.620	.441	.216	.006	-.072	-.204
	.750	.774	.457	.098	.042	.048
	.850	.689	.541	.276	.144	.246
	.950	.363	.331	.227	.264	.288

α = 5.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.085	.452	.164	.329
.053	-.103	.236	-.067	.076
.100	-.195	.064	-.207	-.120
.145	-.134	-.013	-.158	-.063
.189	-.061	.057	-.110	-.025
.234	-.085	.115	-.049	-.038
.280	-.085	.127	.024	-.025
.326	-.085	.121	.000	-.006
.371	-.146	.217	-.097	.051
.392	-.180	.240	-.170	.215
.413	-.219	.267	-.414	.139
.434	-.280	.318	-.657	.127
.457	-.280	.320	-.560	.253
.480	-.329	.340	-.414	.367
.502	-.377	.360	-.304	.392
.551	-.389	.370	-.146	.449
.585	-.414	.376	-.316	.468
.592	-.456	.376	-.450	-.430
.613	-.420	.280	-.463	-.430
.634	-.420	.159	-.499	-.443
.655	-.414	.019	-.505	-.531
.675	-.365	-.166	-.511	-.563
.696	-.341	-.204	-.475	-.601
.774	-.043	-.006	-.195	-.215
.852	-.012	.000	-.018	-.101
.930	.012	-.127	.000	-.114

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.710	.688	.633	.633	.602
	.080	-.038	-.076	-.145	-.201	-.134
	.130	-1.075	-1.261	-1.366	-1.400	-1.436
	.145	-5.926	-5.565	-5.187	-6.000	-5.641
	.155	-2.344	-2.312	-2.309	-2.422	-2.087
	.180	-1.659	-1.414	-1.575	-1.570	-1.722
	.220	-.943	-.942	-1.075	-1.266	-1.193
	.270	-.635	-.713	-.867	-.931	-.949
	.400	-.333	-.331	-.658	-.895	-.949
	.620	.195	.325	-.607	-1.430	-1.448
	.685	-.911	-.408	-1.588	-7.309	-9.652
	.693	-.918	-.656	-2.714	-8.003	-8.788
	.700	-.729	-.662	-1.847	-5.033	-6.639
	.720	-.610	-.535	-.696	-2.240	-2.891
	.750	-.654	-.624	-.633	-1.412	-1.996
	.800	-.553	-.630	-.576	-.864	-1.552
	.900	-.446	-.471	-.671	-.438	-1.120
	.980	-.459	-.471	-.569	-.024	-.426
Lower	.025	.031	.293	.342	.353	.183
	.120	.088	.287	.310	.274	.097
	.220	.421	.280	.266	.274	.085
	.300	.559	.395	.367	.420	.389
	.620	.672	.675	.645	.651	.243
	.750	.811	.739	.746	.706	.602
	.850	.559	.573	.576	.639	.548
	.950	.283	.229	.240	.438	.365

α = 13.3°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.090	.552	-.104	.255
.053	-.256	.395	-.279	.007
.100	-.186	.198	-.383	-.203
.145	-.141	.105	-.318	-.203
.189	-.077	.138	-.260	-.170
.234	-.090	.191	-.110	-.177
.280	-.090	.204	.052	-.190
.326	-.109	.211	.013	-.216
.371	-.237	.316	-.188	-.190
.392	-.270	.370	-.539	-.150
.413	-.308	.435	-.851	.170
.434	-.372	.448	-1.104	.504
.457	-.372	.450	-.851	.595
.480	-.378	.455	-.617	.556
.502	-.449	.460	-.461	.504
.551	-.397	.460	-.266	.536
.585	-.397	.461	-.409	.608
.592	-.436	.435	-.435	-.392
.613	-.353	.296	-.481	-.510
.634	-.372	.198	-.513	-.510
.655	-.391	.020	-.539	-.589
.675	-.314	-.125	-.533	-.589
.696	-.288	-.165	-.487	-.595
.774	-.090	.040	-.175	-.395
.852	-.036	.053	-.039	-.196
.930	-.019	.007	-.026	.013

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.045	-.652	-.896	-1.156	-.385
	.080	-.838	-.830	-.948	-1.085	-.910
	.130	-2.183	-2.338	-2.538	-2.579	-2.628
	.145	-7.978	-7.679	-7.287	-8.374	-7.795
	.155	-3.255	-3.425	-3.526	-3.729	-3.167
	.180	-2.150	-2.127	-2.348	-2.410	-2.481
	.220	-1.260	-1.403	-1.622	-1.852	-1.679
	.270	-.858	-1.041	-1.236	-1.358	-1.359
	.400	-.546	-.487	-.857	-1.176	-1.269
	.620	.013	.178	-.654	-1.897	-1.859
	.685	-.838	-.389	-1.171	-7.712	-11.301
	.693	-.838	-.751	-2.283	-8.524	-10.333
	.700	-.650	-.645	-1.642	-5.269	-7.833
	.720	-.513	-.520	-.720	-2.397	-3.468
	.750	-.513	-.566	-.641	-1.488	-2.410
	.800	-.539	-.586	-.595	-.903	-1.846
	.900	-.487	-.474	-.674	-.481	-1.474
	.980	-.494	-.461	-.608	-.097	-.872
Lower	.025	.461	.659	.628	.708	.526
	.120	.793	.836	.739	.760	.564
	.220	.793	.810	.752	.747	.609
	.300	.643	.705	.661	.663	.519
	.620	.754	.797	.752	.715	.128
	.750	.845	.896	.798	.734	.596
	.850	.604	.659	.563	.630	.468
	.950	.299	.290	.229	.448	.212

TABLE 14 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.7$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	Fuselage, 0.000, Upper surface	Fuselage, 0.000, Lower surface	Fuselage, 0.154, Upper surface	Fuselage, 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron, 0.221	0.426	0.640	0.800	0.918
					$\alpha = 19.0°$						
.032	-.178	.735	-.341	.136	Upper	.010	-3.349	-2.504	-2.787	-3.005	-2.727
.053	-.316	.548	-.481	-.065		.080	-1.573	-1.382	-2.163	-2.497	-1.600
.100	-.198	.314	-.574	-.292		.130	-2.938	-2.804	-2.878	-3.045	-3.332
.145	-.158	.214	-.521	-.331		.145	-9.212	-8.380	-7.634	-8.827	-8.944
.189	-.072	.234	-.461	-.266		.155	-3.886	- .973	-3.957	-4.314	-3.958
.234	-.072	.274	-.187	-.325		.180	-2.515	- .524	-2.722	-2.878	-3.049
.280	-.079	.294	.067	-.331		.220	-1.493	- .689	-1.858	-2.123	-2.114
.326	-.132	.300	.027	-.396		.270	-1.022	- .209	-1.384	-1.603	-1.706
.371	-.323	.407	-.274	-.429		.400	-.565	.568	-.871	-1.222	-1.587
.392	-.375	.047	-.815	-.500		.620	-.087	.007	-.533	-2.023	-1.963
.413	-.428	.528	-1.042	.182		.685	-.854	.381	-.006	-5.021	-10.867
.434	-.501	.548	-1.576	.578		.693	-.861	.608	-1.228	-5.462	-9.852
.457	-.468	.450	-1.215	.676		.700	-.666	.661	-1.027	-3.185	-7.403
.480	-.421	.450	-.835	.617		.720	-.545	.514	-.598	-1.282	-3.168
.502	-.461	.450	-.621	.591		.750	-.598	.614	-.559	-.801	-2.114
.551	-.362	.450	-.347	.559		.800	-.578	.601	-.578	-.621	-1.594
.585	-.342	.474	-.414	.624		.900	-.511	.441	-.630	-.608	-1.166
.592	-.369	.454	-.521	-.435		.980	-.511	-.427	-.546	-.501	-.619
.613	-.290	.341	-.534	-.461	Lower	.025	.693	.815	.780	.815	.665
.634	-.329	.207	-.568	-.513		.120	.881	.875	.780	.755	.586
.655	-.329	.033	-.588	-.630		.220	.841	.835	.799	.775	.619
.675	-.224	-.134	-.561	-.598		.300	.753	.748	.702	.714	.573
.696	-.184	.127	-.487	-.546		.620	.767	.808	.773	.735	.138
.774	-.105	.100	-.200	-.390		.750	.847	.895	.773	.735	.612
.852	-.026	.033	-.087	-.227		.850	.652	.688	.585	.628	.533
.930	[illegible]	.127	-.127	.136		.950	.363	.314	.266	.361	.329
					$\alpha = 23.1°$						
.032	-.269	.803	-.491	.040	Upper	.010	-7.038	- .471	-3.655	-3.580	-3.335
.053	-.356	.619	-.607	-.132		.080	-1.812	- .135	-3.734	-3.546	-1.701
.100	-.235	.389	-.689	-.382		.130	-2.983	- .713	-2.871	-2.912	-2.488
.145	-.215	.310	-.621	-.415		.145	-8.811	- .594	-6.586	-6.615	-6.516
.189	-.067	.336	-.559	-.356		.155	-3.637	- .912	-3.682	-3.444	-2.649
.234	-.087	.356	-.218	-.421		.180	-2.335	- .582	-2.595	-2.394	-2.071
.280	-.128	.375	.075	-.448		.220	-1.380	- .791	-1.778	-1.678	-1.251
.326	-.208	.395	-.102	-.540		.270	-1.125	- .291	-1.317	-1.268	-.921
.371	-.430	.487	-.471	-.599		.400	-.687	.672	-.843	-.880	-.921
.392	-.500	.540	-.941	-.757		.620	-.222	.224	-.724	-.968	-.841
.413	-.572	.586	-1.193	.198		.685	-1.066	-.672	-.263	-2.223	-1.990
.434	-.598	.566	-1.944	.659		.693	-1.079	-.856	-1.581	-2.421	-1.607
.457	-.565	.570	-1.350	.731		.700	-.857	.962	-1.324	-1.562	-1.399
.480	-.511	.560	-1.057	.685		.720	-.713	.883	-.803	-1.002	-.881
.502	-.484	.540	-.818	.652		.750	-.726	.915	-.751	-.982	-.921
.551	-.343	.520	-.552	.593		.800	-.693	.593	-.659	-.934	-.928
.585	-.316	.494	-.696	.632		.900	-.648	.586	-.566	-.880	-.935
.592	-.343	.487	-.839	-.692		.980	-.661	.553	-.540	-.798	-.874
.613	-.262	.342	-.764	-.514	Lower	.025	.785	.876	.856	.859	.706
.634	-.303	.211	-.689	-.652		.120	-.922	.863	.810	.805	.632
.655	-.269	.040	-.648	-.711		.220	-.863	.856	.810	.805	.679
.675	-.175	-.125	-.559	-.632		.300	-.791	.790	.771	.777	.625
.696	-.121	-.099	-.457	-.474		.620	-.791	.830	.810	.750	.323
.774	-.074	.165	-.280	-.320		.750	-.863	.869	.810	.723	.632
.852	-.128	.053	-.102	-.178		.850	-.674	.692	.612	.600	.511
.930	-.114	.171	-.205	.198		.950	-.386	.303	.290	.273	.195

L-764

L-927

TABLE 14 Continued

(c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.6$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = -1.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.290	.288	.297	.286
.053	.060	.072	.059	.052
.100	-.079	-.030	-.083	-.064
.145	-.073	-.072	-.065	-.035
.189	.000	-.012	.012	.006
.234	-.024	.060	-.018	.017
.280	-.024	.084	-.071	.064
.326	.000	.084	-.042	.099
.371	-.024	.126	-.024	.169
.392	-.067	.132	.107	.367
.413	-.054	.138	.006	-.128
.434	-.097	.144	-.154	-.274
.457	-.109	.148	-.172	-.233
.480	-.157	.155	-.095	-.152
.502	-.236	.170	-.012	-.012
.551	-.321	.180	.071	.245
.585	-.375	.192	-.309	.175
.592	-.429	.192	-.445	-.373
.613	-.490	.126	-.493	-.122
.634	-.435	.024	-.493	-.338
.655	-.435	-.066	-.469	-.443
.675	-.363	-.174	-.451	-.501
.696	-.357	-.198	-.421	-.507
.774	-.030	-.036	-.220	-.186
.852	.024	-.048	-.006	-.105
.930	.036	-.126	.018	-.140

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.962	.889	.833	.807	.798
	.080	.569	.493	.420	.362	.381
	.130	-.117	-.264	-.367	-.410	-.502
	.145	-3.459	-3.179	-2.925	-3.692	-3.562
	.155	-1.137	-1.100	-1.066	-1.217	-1.028
	.180	-.792	-.601	-.746	-.783	-.980
	.220	-.346	-.349	-.460	-.706	-.641
	.270	-.199	-.234	-.350	-.457	-.496
	.400	-.006	.030	-.274	-.546	-.538
	.620	.621	.673	-.390	-.819	-.895
	.685	-.862	-.397	-1.200	-5.835	-5.957
	.693	-.891	-.679	-2.214	-6.369	-5.128
	.700	-.704	-.661	-1.597	-3.811	-3.544
	.720	-.604	-.511	-.606	-1.525	-1.125
	.750	-.621	-.553	-.554	-.914	-.768
	.800	-.451	-.595	-.524	-.510	-.677
	.900	-.358	-.463	-.554	-.273	-.629
	.980	-.358	-.433	-.490	-.047	-.460
Lower	.025	-.264	-.246		.047	-.054
	.120	-.334	-.174	-.017	.006	-.073
	.220	-.322	-.204	-.029	-.012	-.060
	.300	-.223	-.252	-.082	-.030	-.073
	.620	.387	.204	-.052	-.131	-.254
	.750	.715	.427	.029	-.018	-.024
	.850	.680	.535	.216	.077	.133
	.950	.358	.325	.198	.214	.230

$\alpha = 5.9°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.080	.465	.152	.325
.053	-.104	.233	-.067	.073
.100	-.196	.073	-.183	-.098
.145	-.147	-.006	-.158	-.067
.189	-.086	.055	-.085	-.037
.234	-.104	.116	-.049	-.037
.280	-.092	.135	.012	-.024
.326	-.012	.135	.000	.006
.371	-.159	.184	-.079	.061
.392	-.180	.210	-.146	.090
.413	-.220	.257	-.359	.122
.434	-.257	.294	-.596	.122
.457	-.269	.310	-.499	.257
.480	-.300	.330	-.304	.349
.502	-.367	.360	-.195	.392
.551	-.398	.380	-.116	.441
.585	-.441	.361	-.329	.459
.592	-.521	.349	-.487	-.484
.613	-.551	.251	-.511	-.478
.634	-.570	.129	-.487	-.484
.655	-.600	-.018	-.505	-.551
.675	-.527	-.171	-.596	-.576
.696	-.508	-.220	-.615	-.576
.774	-.092	-.031	-.383	-.245
.852	-.024	.006	-.049	-.104
.930	.018	-.116	-.024	-.104

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.727	.723	.661	.651	.612
	.080	.038	.037	-.049	-.158	-.135
	.130	-.974	-1.041	-1.200	-1.302	-1.402
	.145	-5.560	-4.912	-4.734	-5.672	-5.561
	.155	-2.157	-1.990	-2.039	-2.246	-2.009
	.180	-1.531	-1.176	-1.378	-1.467	-1.684
	.220	-.822	-.735	-.906	-1.175	-1.158
	.270	-.512	-.508	-.698	-.846	-.913
	.400	-.152	-.122	-.465	-.803	-.925
	.620	.607	.612	-.459	-1.363	-1.427
	.685	-.854	-.423	-1.115	-6.810	-9.683
	.693	-.860	-.686	-2.229	-7.583	-8.862
	.700	-.683	-.704	-1.635	-4.674	-6.670
	.720	-.531	-.527	-.674	-2.033	-2.842
	.750	-.519	-.563	-.606	-1.254	-1.941
	.800	-.506	-.570	-.551	-.749	-1.482
	.900	-.519	-.521	-.606	-.402	-1.066
	.980	-.544	-.514	-.576	-.055	-.355
Lower	.025	-.006	.269	.312	.323	.165
	.120	.063	.257	.294	.286	.110
	.220	.399	.263	.263	.237	.080
	.300	.557	.355	.325	.347	.349
	.620	.671	.643	.606	.639	.251
	.750	.803	.729	.692	.682	.582
	.850	.557	.563	.563	.627	.563
	.950	.266	.208	.220	.438	.380

$\alpha = 13.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.056	.643	-.095	.256
.053	-.237	.437	-.272	.051
.100	-.175	.212	-.367	-.186
.145	-.131	.112	-.316	-.186
.189	-.062	.162	-.253	-.128
.234	-.081	.200	-.108	-.167
.280	-.087	.225	.063	-.167
.326	-.094	.225	.025	-.173
.371	-.200	.306	-.190	-.167
.392	-.240	.350	-.487	-.103
.413	-.287	.406	-.772	.205
.434	-.343	.450	-.999	.513
.457	-.325	.455	-.740	.596
.480	-.350	.460	-.500	.551
.502	-.393	.455	-.329	.506
.551	-.375	.420	-.342	.526
.585	-.400	.412	-.304	.596
.592	-.487	.412	-.506	-.487
.613	-.406	.306	-.519	-.455
.634	-.468	.181	-.512	-.481
.655	-.481	.025	-.544	-.551
.675	-.462	-.150	-.576	-.551
.696	-.481	-.162	-.620	-.577
.774	-.200	.050	-.335	-.288
.852	-.012	.044	-.095	-.192
.930	.000	-.012	-.051	.032

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.075	-.112	-.474	-.905	-.162
	.080	-.649	-.649	-.821	-.987	-.787
	.130	-1.911	-2.035	-2.288	-2.391	-2.448
	.145	-7.305	-6.837	-6.705	-7.844	-7.486
	.155	-2.910	-2.972	-3.154	-3.454	-2.972
	.180	-1.911	-1.823	-2.083	-2.201	-2.323
	.220	-1.055	-1.161	-1.385	-1.657	-1.573
	.270	-.668	-.812	-1.038	-1.215	-1.236
	.400	-.331	-.225	-.647	-1.044	-1.161
	.620	.462	.487	-.538	-1.816	-1.786
	.685	-.843	-.425	-.833	-7.085	-11.151
	.693	-.843	-.637	-1.968	-7.901	-10.265
	.700	-.649	-.687	-1.462	-4.770	-7.780
	.720	-.524	-.524	-.647	-2.138	-3.415
	.750	-.512	-.524	-.577	-1.303	-2.341
	.800	-.493	-.537	-.551	-.765	-1.792
	.900	-.531	-.537	-.603	-.424	-1.349
	.980	-.531	-.512	-.558	-.133	
Lower	.025	.437	.637	.635	.658	.531
	.120	.762	.799	.744	.746	.574
	.220	.755	.774	.744	.740	.612
	.300	.624	.687	.673	.633	.524
	.620	.712	.780	.756	.709	.150
	.750	.805	.868	.782	.734	.606
	.850	.562	.643	.564	.620	.493
	.950	.300	.256	.231	.418	.287

TABLE 14 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 5.6 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = 18.4°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.175	.715	-.299	.124
.053	-.293	.538	-.452	-.078
.100	-.194	.310	-.509	-.301
.145	-.162	.221	-.452	-.314
.189	-.081	.247	-.395	-.268
.234	-.062	.297	-.172	-.314
.280	-.087	.297	.057	-.314
.326	-.106	.323	.025	-.392
.371	-.287	.424	-.223	-.412
.392	-.330	.475	-.707	-.451
.413	-.381	.525	-.949	.183
.434	-.431	.550	-1.401	.589
.457	-.400	.550	-1.070	.661
.480	-.387	.550	-.669	.634
.502	-.418	.530	-.458	.589
.551	-.350	.510	-.293	.569
.585	-.362	.487	-.331	.621
.592	-.406	.468	-.503	-.477
.613	-.350	.348	-.522	-.471
.634	-.400	.209	-.535	-.523
.655	-.421	.051	-.560	-.595
.675	-.350	-.114	-.579	-.595
.696	-.316	-.108	-.592	-.602
.774	-.187	.128	-.287	-.262
.852	-.062	.057	-.108	-.281
.930	-.037	.152	-.115	.144

Surface	x/c	0.221	0.4[illegible]6	0.640	0.800	0.918
Upper	.010	-3.171	-2.170	-2.571	-2.674	-2.354
	.080	-1.439	-1.151	-1.779	-2.120	-1.324
	.130	-2.681	-2.5[illegible]1	-2.845	-2.834	-3.053
	.145	-8.463	-7.534	-7.476	-8.253	-8.335
	.155	-3.496	-3.5[illegible]7	-3.787	-3.967	-3.565
	.180	-2.248	-2.189	-2.551	-2.611	-2.778
	.220	-1.242	-1.4[illegible]1	-1.720	-1.917	-1.898
	.270	-.809	-.9[illegible]2	-1.256	-1.414	-1.499
	.400	-.388	-.2[illegible]7	-.667	-1.083	-1.399
	.620	.287	.2[illegible]9	-.700	-1.802	-1.848
	.685	-.860	-.3[illegible]8	-.026	-4.400	-10.433
	.693	-.826	-.6[illegible]7	-1.328	-4.776	-9.491
	.700	-.630	-.6[illegible]6	-1.092	-2.713	-7.143
	.720	-.497	-.4[illegible]1	-.641	-1.019	-3.022
	.750	-.497	-.417	-.615	-.681	-1.998
	.800	-.490	-.417	-.628	-.592	-1.461
	.900	-.529	-.417	-.674	-.579	-1.043
	.980	-.503	-.4[illegible]4	-.615	-.490	-.475
Lower	.025	.681	.803	.791	.796	.624
	.120	.879	.854	.785	.745	.549
	.220	.841	.816	.798	.777	.593
	.300	.732	.753	.693	.694	.531
	.620	.758	.8[illegible]9	.778	.726	.125
	.750	.860	.8[illegible]9	.785	.726	.574
	.850	.630	.6[illegible]4	.576	.630	.500
	.950	.363	.3[illegible]0	.262	.450	.312

α = 23.1°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.244	.874	-.468	.032
.053	-.343	.615	-.572	-.136
.100	-.200	.419	-.624	-.377
.145	-.150	.321	-.585	-.429
.189	-.081	.321	-.500	-.364
.234	-.062	.353	-.201	-.416
.280	-.106	.386	.084	-.442
.326	-.181	.392	-.013	-.520
.371	-.393	.491	-.409	-.565
.392	-.465	.545	-.858	-.689
.413	-.531	.678	-1.091	.240
.434	-.568	.621	-1.787	.637
.457	-.570	.620	-1.182	.721
.480	-.462	.590	-.916	.689
.502	-.443	.550	-.663	.637
.551	-.325	.520	-.396	.591
.585	-.318	.517	-.520	.637
.592	-.356	.484	-.643	-.624
.613	-.281	.340	-.669	-.572
.634	-.331	.190	-.676	-.663
.655	-.325	.013	-.702	-.754
.675	-.237	-.164	-.663	-.715
.696	-.181	-.144	-.565	-.663
.774	-.119	.131	-.110	-.450
.852	-.069	.059	-.071	-.240
.930	-.081	.177	-.208	.208

Surface	x/c	0.221	0.4[illegible]6	0.640	0.800	0.918
Upper	.010	-6.340	-3.2[illegible]8	-3.447	-3.281	-3.047
	.080	-1.686	-2.8[illegible]1	-3.476	-3.248	-1.499
	.130	-2.814	-2.5[illegible]7	-2.677	-2.781	-2.423
	.145	-8.276	-7.2[illegible]8	-6.406	-6.503	-6.456
	.155	-3.378	-3.6[illegible]3	-3.508	-3.365	-2.604
	.180	-2.160	-2.3[illegible]1	-2.443	-2.313	-2.042
	.220	-1.301	-1.5[illegible]3	-1.624	-1.611	-1.255
	.270	-1.051	-1.0[illegible]9	-1.143	-1.169	-.924
	.400	-.596	-.4[illegible]1	-.637	-.780	-.980
	.620	-.006	.0[illegible]7	-.676	-.916	-.874
	.685	-1.032	-.5[illegible]9	.013	-2.170	-1.792
	.693	-1.045	-.7[illegible]5	-1.351	-2.443	-1.474
	.700	-.833	-.9[illegible]6	-1.163	-1.358	-1.317
	.720	-.679	-.8[illegible]8	-.715	-.682	-.843
	.750	-.699	-.9[illegible]9	-.721	-.715	-.887
	.800	-.712	-.6[illegible]1	-.721	-.741	-.887
	.900	-.647	-.5[illegible]6	-.591	-.708	-.887
	.980	-.673	-.5[illegible]7	-.546	-.585	-.824
Lower	.025	.788	.8[illegible]7	.838	.858	.693
	.120	.910	.8[illegible]7	.793	.793	.631
	.220	.878	.8[illegible]7	.819	.793	.674
	.300	.776	.7[illegible]2	.741	.747	.618
	.620	.776	.8[illegible]4	.780	.728	.300
	.750	.853	.8[illegible]3	.786	.741	.624
	.850	.635	.6[illegible]7	.598	.617	.487
	.950	.340	.3[illegible]1	.286	.292	.194

L-927

L-927

TABLE 14 Continued
(d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 8.4$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

x/l	0.000, Upper surface (Fuselage)	0.000, Lower surface (Fuselage)	0.154, Upper surface (Fuselage)	0.154, Lower surface (Fuselage)	Surface	x/c	0.221	0.426	0.640	0.800	0.918
α = −1.2°											
.032	.285	.309	.292	.305	Upper	.010	.950	.690	.850	.822	.789
.053	.089	.077	.055	.076		.080	.608	.508	.469	.396	.415
.100	−.053	−.059	−.091	−.070		.130	−.035	−.166	−.246	−.371	−.439
.145	−.053	−.071	−.067	−.029		.145	−4.221	−2.613	−2.662	−3.572	−3.583
.189	.012	−.018	.005	.012		.155	−.973	−.567	−.891	−1.132	−.908
.234	−.012	.053	−.030	.029		.180	−.684	−.451	−.592	−.712	−.914
.280	−.012	.071	−.067	.059		.220	−.236	−.202	−.328	−.639	−.582
.326	.024	.077	−.030	.094		.270	−.071	−.071	−.211	−.408	−.457
.371	−.006	.113	.018	.182		.400	.218	.261	−.100	−.499	−.475
.392	−.059	−.047	.116	.387		.620	1.068	.878	−.364	−.822	−.843
.413	−.036	.119	.037	−.176		.685	−.820	−.427	−.281	−5.903	−6.191
.434	−.065	.131	−.128	−.322		.693	−.808	−.659	−1.360	−6.238	−5.366
.457	−.071	.135	−.103	−.293		.700	−.537	−.694	−1.114	−3.804	−3.716
.480	−.131	.150	−.006	−.205		.720	−.519	−.546	−.592	−1.442	−1.140
.502	−.190	.175	.116	−.047		.750	−.531	−.570	−.551	−.852	−.694
.551	−.309	.190	−.055	.205		.800	−.454	−.568	−.551	−.523	−.623
.585	−.392	.196	−.456	.193		.900	−.407	−.505	−.586	−.259	−.576
.592	−.487	.196	−.511	−.364		.980	−.437	−.461	−.492	−.195	−.404
.613	−.552	.131	−.511	−.158	Lower	.025	−.307	−.279	−.059	.012	−.071
.634	−.510	.012	−.499	−.410		.120	−.395	−.214	−.064	−.024	−.089
.655	−.510	−.071	−.517	−.510		.220	−.366	−.214	−.064	−.043	−.089
.675	−.457	−.196	−.529	−.551		.300	−.242	−.285	−.117	−.085	−.119
.696	−.463	−.214	−.505	−.528		.620	.366	.131	−.117	−.110	−.297
.774	−.047	−.042	−.268	−.240		.750	.708	.474	−.018	−.024	−.074
.852	.059	−.059	−.018	−.088		.850	.696	.510	.129	.055	.154
.930	.065	−.160	.037	−.111		.950	.354	.532	.141	.164	.261
α = 6.0°											
.032	.103	.478	.160	.345	Upper	.010	.790	.753	.696	.703	.657
.053	−.097	.239	−.049	.092		.080	.116	.073	.006	−.074	−.037
.100	−.164	.067	−.197	−.074		.130	−.836	−.919	−1.048	−1.171	−1.235
.145	−.116	−.006	−.160	−.062		.145	−5.120	−4.544	−4.364	−5.393	−5.197
.189	−.055	.049	−.080	−.012		.155	−1.905	−1.764	−1.824	−2.108	−1.620
.234	−.067	.110	−.031	−.018		.180	−1.323	−1.011	−1.208	−1.338	−1.576
.280	−.061	.141	.025	−.006		.220	−.637	−.588	−.764	−1.085	−1.053
.326	−.055	.141	.018	.006		.270	−.367	−.343	−.549	−.746	−.846
.371	−.097	.154	−.055	.092		.400	.092	.141	−.296	−.727	−.852
.392	.043	.220	−.111	.271		.620	1.035	.876	−.518	−1.301	−1.345
.413	−.158	.269	−.308	.155		.685	−.882	−.478	−.321	−6.521	−9.360
.434	−.219	.294	−.530	.111		.693	−.876	−.729	−1.510	−7.267	−8.569
.457	−.219	.305	−.425	.222		.700	−.704	−.759	−1.207	−4.413	−6.420
.480	−.250	.320	−.210	.333		.720	−.533	−.594	−.647	−1.843	−2.659
.502	−.292	.340	−.012	.382		.750	−.502	−.606	−.635	−1.128	−1.783
.551	−.359	.360	−.148	.431		.800	−.484	−.606	−.616	−.703	−1.331
.585	−.463	.380	−.481	.456		.900	−.545	−.606	−.635	−.481	−.931
.592	−.578	.343	−.573	−.542		.980	−.563	−.563	−.586	−.222	−.298
.613	−.730	.239	−.518	−.481	Lower	.025	.012	.233	.308	.327	.164
.634	−.627	.110	−.512	−.487		.120	.067	.220	.296	.271	.055
.655	−.627	−.043	−.579	−.567		.220	.325	.208	.259	.259	.061
.675	−.596	−.196	−.629	−.616		.300	.539	.276	.277	.296	.201
.696	−.615	−.227	−.641	−.610		.620	.655	.661	.579	.616	.286
.774	−.152	−.018	−.413	−.314		.750	.784	.741	.641	.666	.584
.852	.055	.024	−.049	−.111		.850	.545	.576	.542	.647	.566
.930	.055	−.092	.025	−.062		.950	.257	.208	.210	.431	.426
α = 13.4°											
.032	−.078	.641	−.087	.250	Upper	.010	.133	.044	−.206	−.397	−.123
.053	−.253	.190	−.273	.025		.080	−.620	−.566	−.718	−.837	−.806
.100	−.195	.201	−.141	−.181		.130	−1.797	−1.848	−2.078	−2.171	−2.501
.145	−.149	.094	−.285	−.194		.145	−6.990	−6.410	6.169	7.308	−7.595
.189	−.091	.132	−.236	−.125		.155	−2.758	−2.702	−2.853	−3.139	−3.054
.234	−.097	.182	−.081	−.169		.180	−1.765	−1.609	−1.854	−1.998	−2.358
.280	−.097	.220	.043	−.150		.220	−.911	−.968	−1.199	−1.501	−1.605
.326	−.084	.220	.031	−.200		.270	−.557	−.591	−.862	−1.073	−1.267
.371	−.208	.295	−.130	−.169		.400	−.183	.069	−.412	−.926	−1.176
.392	−.355	.355	−.407	−.100		.620	.867	.760	−.737	−1.681	−1.806
.413	−.375	.421	−.670	.200		.685	−.943	−.459	−.250	−6.297	−11.311
.434	−.338	.446	−.856	.487		.693	−.911	−.767	−1.467	−7.084	−10.362
.457	−.318	.440	−.620	.587		.700	−.727	−.742	−1.199	−4.181	−7.887
.480	−.331	.435	−.441	.537		.720	−.569	−.578	−.656	−1.749	−3.398
.502	−.377	.430	−.136	.512		.750	−.550	−.578	−.631	−1.040	−2.319
.551	−.409	.428	−.223	.512		.800	−.538	−.584	−.637	−.608	−1.715
.585	−.416	.427	−.484	.587		.900	−.582	−.572	−.674	−.533	−1.234
.592	−.487	.420	−.558	−.518		.980	−.582	−.559	−.618	−.354	−.513
.613	−.513	.289	−.521	−.524	Lower	.025	.405	.616	.606	.645	.513
.634	−.611	.157	−.527	−.549		.120	.753	.811	.724	.738	.565
.655	−.643	−.006	−.559	−.624		.220	.772	.779	.731	.726	.630
.675	−.617	−.170	−.602	−.618		.300	.633	.591	.649	.658	.500
.696	−.591	−.195	−.602	−.618		.620	.727	.811	.737	.707	.136
.774	−.292	.031	−.428	−.300		.750	.829	.874	.787	.732	.617
.852	.026	.044	−.130	−.219		.850	.582	.628	.562	.633	.513
.930	.026	−.031	−.081	[illegible]		.950	.304	.245	.187	.385	.318

TABLE 14 Continued
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 8.4 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					α = 19.0°						
.032	-.172	.739	-.295	.121	Upper	.010	-2.154	-1.999	-2.299	-2.407	-2.241
.053	-.312	.522	-.440	-.070		.080	-1.256	-1.032	-1.452	-1.653	-1.356
.100	-.197	.331	-.496	-.293		.130	-2.442	-2.458	-2.643	-2.684	-3.012
.145	-.146	.210	-.452	-.312		.145	-7.840	-7.285	-6.941	-7.912	-8.183
.189	-.064	.248	-.383	-.267		.155	-3.160	-3.318	-3.420	-3.695	-3.502
.234	-.057	.293	-.145	-.299		.180	-2.013	-2.044	-2.267	-2.420	-2.745
.280	-.083	.299	.044	-.306		.220	-1.058	-1.261	-1.484	-1.741	-1.840
.326	-.115	.312	.025	-.382		.270	-.635	-.821	-1.044	-1.263	-1.471
.371	-.280	.408	-.214	-.388		.400	-.321	-.013	-.439	-.924	-1.337
.392	-.330	.470	-.641	-.408		.620	.538	.312	-.771	-1.697	-1.764
.413	-.388	.535	-.855	.204		.685	-.859	-.465	.076	-3.689	-10.157
.434	-.427	.548	-1.307	.599		.693	-.821	-.694	-1.242	-3.890	-9.208
.457	-.427	.540	-.911	.662		.700	-.667	-.739	-1.063	-2.130	-6.903
.480	-.395	.520	-.547	.624		.720	-.500	-.560	-.624	-.792	-2.859
.502	-.427	.500	-.308	.579		.750	-.506	-.560	-.630	-.666	-1.828
.551	-.382	.480	-.371	.535		.800	-.506	-.567	-.630	-.641	-1.254
.585	-.388	.471	-.515	.618		.900	-.532	-.586	-.681	-.610	-.841
.592	-.439	.465	-.547	-.516		.980	-.538	-.560	-.637	-.515	-.376
.613	-.401	.318	-.534	-.484	Lower	.025	.673	.815	.777	.786	.618
.634	-.503	.185	-.534	-.567		.120	.885	.866	.777	.760	.535
.655	-.560	.019	-.572	-.624		.220	.846	.815	.783	.760	.592
.675	-.478	-.159	-.610	-.611		.300	.744	.739	.688	.710	.554
.696	-.465	-.166	-.610	-.630		.620	.776	.802	.751	.710	.146
.774	-.217	.076	-.321	-.434		.750	.878	.879	.751	.723	.592
.852	-.083	.038	-.151	-.318		.850	.641	.669	.579	.610	.529
.930	-.038	.127	-.138	.166		.950	.372	.280	.236	.333	.357
					α = 23.2°						
.032	-.223	.796	-.427		Upper	.010	-5.933	-2.942	-3.259	-3.299	-3.158
.053	-.344	.611	-.509	-.136		.080	-1.533	-2.541	-3.259	-3.292	-1.968
.100	-.223	.388	-.599	-.394		.130	-2.451	-2.369	-2.536	-2.725	-2.630
.145	-.178	.299	-.548	-.419		.145	-7.051	-6.642	-6.260	-6.642	-7.043
.189	-.070	.331	-.478	-.355		.155	-2.778	-3.305	-3.317	-3.400	-2.999
.234	-.045	.357	-.159	-.407		.180	-1.760	-2.114	-2.272	-2.318	-2.350
.280	-.083	.357	.089	-.413		.220	-1.207	-1.337	-1.478	-1.643	-1.547
.326	-.197	.369	-.051	-.490		.270	-1.087	-.828	-1.013	-1.191	-1.197
.371	-.369	.484	-.369	-.516		.400	-.459	-.178	-.478	-.777	-1.350
.392	-.460	.530	-.732	-.626		.620	.207	.038	-.555	-1.223	-1.140
.413	-.535	.586	-.955	.239		.685	-.936	-.503	.252	-1.694	-2.375
.434	-.554	.592	-1.547	.652		.693	-.905	-.713	-1.181	-1.789	-2.031
.457	-.503	.590	-1.089	.723		.700	-.748	-.790	-1.000	-1.242	-1.770
.480	-.458	.550	-.860	.684		.720	-.597	-.662	-.613	-.802	-1.102
.502	-.452	.530	-.503	.652		.750	-.597	-.707	-.626	-.777	-1.063
.551	-.363	.505	-.388	.581		.800	-.603	-.586	-.665	-.783	-.993
.585	-.395	.484	-.554	.652		.900	-.584	-.516	-.665	-.707	-.879
.592	-.490	.471	-.605	-.574		.980	-.578	-.497	-.607	-.650	-.745
.613	-.382	.331	-.630	-.490	Lower	.025	.767	.866	.845	.834	.694
.634	-.484	.178	-.662	-.568		.120	.905	.860	.807	.758	.630
.655	-.458	.025	-.681	-.691		.220	.867	.834	.845	.802	.688
.675	-.369	-.172	-.662	-.691		.300	.779	.783	.768	.745	.630
.696	-.318	-.153	-.599	-.716		.620	.779	.828	.807	.739	.287
.774	-.172	.115	-.204	-.500		.750	.861	.872	.794	.726	.630
.852	-.070	.038	-.076	-.303		.850	.660	.675	.607	.611	.516
.930	-.070	.185	-.166	.200		.950	.390	.293	.265	.293	.242

L-927

L-927

TABLE 14 Continued

(e)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.7$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = -1.6°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
.032	.273	.275	.276	.296	Upper	.010	.924	.830	.810	.815	.825
.053	.031	.031	.049	.054		.080	.416	.287	.266	.220	.304
.100	-.081	-.056	-.116	-.091		.130	-.378	-.681	-.768	-.772	-.726
.145	-.081	-.094	-.073	-.042		.145	-4.473	-4.452	-4.173	-4.942	-4.311
.189	-.006	-.044	-.018	-.006		.155	-1.669	-1.817	-1.784	-1.813	-1.371
.234	-.050	.025	-.043	.012		.180	-1.247	-1.199	-1.264	-1.249	-1.259
.280	-.043	.056	-.049	.036		.220	-.757	-.880	-.925	-1.066	-.850
.326	-.037	.062	-.037	.079		.270	-.602	-.787	-.841	-.815	-.670
.371	-.118	.112	-.147	.139		.400	-.577	-.749	-.895	-.919	-.726
.392	-.031	.125	.012	.357		.620	-.974	-1.036	-1.379	-1.164	-1.141
.413	-.155	.144	-.147	.024		.685	-3.288	-2.054	-5.400	-7.147	-5.565
.434	-.217	.169	-.380	-.133		.693	-3.288	-1.886	-6.930	-7.705	-4.646
.457	-.236	.170	-.484	-.091		.700	-2.308	-1.486	-5.219	-4.771	-3.431
.480	-.304	.190	-.429	.006		.720	-1.303	-.656	-2.050	-2.168	-1.402
.502	-.378	.210	-.447	.097		.750	-.856	-.481	-1.167	-1.390	-1.098
.551	-.416	.230	-.600	.296		.800	-.527	-.524	-.587	-.827	-.955
.585	-.403	.256	-.766	.242		.900	-.347	-.412	-.218	-.404	-.800
.592	-.372	.256	-.827	-.774		.980	-.273	-.331	.121	-.086	-.620
.613	-.310	.212	-.637	-.278							
.634	-.248	.156	-.416	-.726	Lower	.025	-.136	-.050	.115	.147	.031
.655	-.205	.100	-.300	-.314		.120	-.230	-.019	.103	.080	-.006
.675	-.130	.019	-.196	-.115		.220	-.199	-.044	.079	.055	-.006
.696	-.056	.019	-.116	-.048		.300	-.031	-.062	.006	.024	-.068
.774	-.025	.056	-.018	.012		.620	.453	.262	.151	.098	-.199
.852	-.031	-.044	.000	-.121		.750	.707	.437	.248	.196	.180
.930	.062	-.175	.080	-.175		.850	.689	.543	.399	.318	.316
						.950	.453	.393	.472	.367	.292

$\alpha = 5.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
.032	.090	.487	.141	.336	Upper	.010	.637	.614	.549	.564	.596
.053	-.096	.247	-.083	.103		.080	-.086	-.152	-.271	-.282	-.212
.100	-.192	.082	-.205	-.123		.130	-1.164	-1.411	-1.646	-1.622	-1.635
.145	-.135	.000	-.154	-.084		.145	-6.137	-6.067	-5.943	-6.795	-6.269
.189	-.071	.063	-.115	-.039		.155	-2.462	-2.606	-2.756	-2.769	-2.372
.234	-.103	.120	-.051	-.032		.180	-1.770	-1.651	-1.910	-1.827	-1.974
.280	-.083	.127	.013	-.039		.220	-1.060	-1.151	-1.362	-1.500	-1.365
.326	-.083	.145	.019	-.026		.270	-.784	-.955	-1.116	-1.141	-1.122
.371	-.186	.221	-.154	.013		.400	-.582	-.696	-.962	-1.147	-1.135
.392	-.225	.250	-.218	.181		.620	-.429	-.449	-1.007	-1.673	-1.705
.413	-.276	.285	-.494	.187		.685	-1.109	-.911	-3.220	-8.237	-10.628
.434	-.340	.329	-.769	.187		.693	-1.029	-1.082	-4.485	-9.167	-9.673
.457	-.346	.335	-.679	.323		.700	-.766	-.981	-3.175	-5.699	-7.372
.480	-.385	.360	-.609	.413		.720	-.563	-.727	-.942	-2.692	-3.321
.502	-.468	.385	-.526	.445		.750	-.465	-.474	-.697	-1.712	-2.372
.551	-.436	.405	-.455	.478		.800	-.410	-.436	-.671	-1.026	-1.878
.585	-.429	.411	-.487	.510		.900	-.496	-.506	-.594	-.436	-1.365
.592	-.429	.405	-.494	-.465		.980	-.514	-.436	-.652	.026	-.545
.613	-.359	.285	-.423	-.310							
.634	-.340	.183	-.391	-.419	Lower	.025	.092	.348	.381	.378	.192
.655	-.301	.038	-.365	-.542		.120	.141	.297	.336	.288	.083
.675	-.224	-.089	-.308	-.445		.220	.441	.361	.361	.282	.135
.696	-.173	-.095	-.205	-.252		.300	.570	.493	.555	.474	.442
.774	.045	.101	.038	.052		.620	.674	.677	.678	.679	.205
.852	-.032	.051	-.026	-.058		.750	.802	.740	.761	.712	.596
.930	.013	-.101	.013	-.084		.850	.576	.595	.600	.641	.538
						.950	.294	.266	.277	.487	.327

$\alpha = 13.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
.032	-.076	.660	-.084	.267	Upper	.010	-.156	-1.194	-1.401	-1.517	-1.208
.053	-.247	.427	-.265	.070		.080	-.905	-.962	-1.070	-1.233	-1.082
.100	-.202	.214	-.355	-.159		.130	-2.266	-2.501	-2.789	-2.878	-2.935
.145	-.139	.132	-.329	-.166		.145	-8.111	-8.038	-7.807	-9.118	-8.521
.189	-.076	.170	-.265	-.127		.155	-3.390	-3.670	-3.833	-4.098	-3.511
.234	-.095	.195	-.110	-.153		.180	-2.273	-2.325	-2.585	-2.717	-2.752
.280	-.108	.220	.071	-.153		.220	-1.424	-1.640	-1.840	-2.072	-1.891
.326	-.133	.239	-.006	-.178		.270	-1.036	-1.276	-1.445	-1.581	-1.512
.371	-.247	.333	-.239	-.178		.400	-.780	-.867	-1.121	-1.381	-1.449
.392	-.305	.400	-.568	-.146		.620	-.581	-.540	-1.025	-2.097	-2.062
.413	-.361	.452	-.910	.204		.685	-1.161	-.861	-2.483	-9.080	-11.766
.434	-.411	.503	-1.194	.529		.693	-1.043	-.987	-3.566	-9.673	-10.779
.457	-.411	.500	-.974	.611		.700	-.793	-.874	-2.407	-6.369	-8.211
.480	-.436	.490	-.813	.579		.720	-.581	-.635	-.758	-2.988	-3.732
.502	-.512	.485	-.703	.548		.750	-.500	-.440	-.650	-1.930	-2.625
.551	-.455	.480	-.613	.548		.800	-.443	-.390	-.554	-1.297	-2.107
.585	-.424	.478	-.607	.618		.900	-.481	-.440	-.497	-.684	-1.727
.592	-.411	.478	-.587	-.560		.980	-.556	-.390	-.548	-.207	-1.120
.613	-.329	.327	-.497	-.363							
.634	-.297	.226	-.445	-.427	Lower	.025	.462	.679	.681	.697	.550
.655	-.272	.063	-.419	-.497		.120	.812	.823	.771	.710	.544
.675	-.177	-.044	-.355	-.382		.220	.780	.792	.751	.723	.588
.696	-.133	-.025	-.252	-.217		.300	.656	.698	.700	.639	.500
.774	.057	.195	.052	.070		.620	.724	.792	.777	.703	.095
.852	-.044	.101	-.013	-.064		.750	.830	.880	.751	.691	.538
.930	-.006	.050	-.006	.057		.850	.587	.654	.611	.626	.436
						.950	.331	.327	.299	.419	.177

TABLE 14 Concluded
(e) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.7$ $h_d/c = 0.0$

$C_{\mu,k} = 0.012$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	Fuselage: C_p values for spanwise stations, $\frac{y}{b/2}$, of: 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface	Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
$\alpha = 15.8°$											
.032	-.154	.718	-.299	.115	Upper	.010	-4.375	-3.445	-3.502	-3.617	-3.410
.053	-.295	.506	-.427	-.070		.080	-1.751	-2.115	-3.299	-3.515	-2.167
.100	-.212	.295	-.523	-.287		.130	-3.114	-2.923	-3.146	-3.248	-3.474
.145	-.160	.218	-.458	-.306		.145	-9.590	-8.705	-8.157	-8.832	-9.494
.189	-.293	.231	-.395	-.261		.155	-4.139	-4.333	-4.457	-4.553	-4.269
.234	-.090	.276	-.159	-.325		.180	-2.732	-2.872	-3.209	-3.197	-3.372
.281	-.115	.295	.108	-.337		.220	-1.675	-2.026	-2.305	-2.426	-2.385
.326	-.162	.421	.251	-.414		.270	-1.254	-1.583	-1.866	-1.917	-1.974
.371	-.358	.410	-.267	-.458		.400	-.898	-1.096	-1.426	-1.592	-1.891
.392	-.380	.475	-.828	-.573		.620	-.790	-.917	-1.407	-2.254	-2.269
.413	-.455	.538	-1.070	.166		.685	-2.017	-1.436	-2.547	-6.775	-11.673
.434	-.513	.551	-1.575	.599		.693	-1.840	-1.122	-3.695	-7.399	-10.615
.457	-.500	.550	-1.286	.675		.700	-1.299	-.872	-2.655	-4.725	-5.051
.481	-.474	.540	-1.017	.643		.720	-.771	-.474	-.942	-2.337	-3.641
.502	-.526	.543	-.546	.605		.750	-.605	-.372	-.611	-1.522	-2.590
.551	-.414	.525	-.860	.586		.800	-.516	-.394	-.471	-.930	-2.096
.585	-.372	.519	-.593	.650		.900	-.439	-.391	-.369	-.458	-1.660
.592	-.340	.494	-1.274	-1.165		.980	-.363	-.378	-.369	-.185	-1.051
.613	-.256	.385	-.955	-.750	Lower	.025	.713	.621	.790	.821	.622
.634	-.224	.288	-.592	-.484		.120	.898	.621	.720	.739	.513
.655	-.235	.141	-.344	-.363		.220	.847	.708	.777	.777	.571
.676	-.115	.132	-.166	-.191		.300	.739	.744	.707	.688	.506
.696	-.258	.132	-.076	-.064		.620	.771	.801	.751	.726	-.013
.774	.240	.132	.070	.115		.750	.866	.659	.739	.713	.526
.852	-.264	.090	-.206	[illegible]		.850	.650	.692	.624	.650	.462
.877	-.214	.109	.206	.189		.950	.427	.559	.337	.465	.218
$\alpha = 23.[illegible]°$											
.032	-.232	.745	-.424	.038	Upper	.010	-8.535	-3.878	-4.094	-3.865	-3.698
.053	-.342	.535	-.532	-.115		.080	-1.967	-3.623	-4.203	-3.865	-2.401
.100	-.213	.388	-.621	-.337		.130	-3.216	-2.878	-3.273	-3.235	-2.426
.145	-.187	.293	-.531	-.408		.145	-9.322	-7.813	-6.621	-6.269	-6.557
.189	-.232	.282	-.493	-.325		.155	-3.952	-4.228	-3.929	-3.466	-2.749
.234	-.077	.249	-.183	-.388		.180	-2.566	-2.123	-2.885	-2.556	-2.194
.281	-.129	.337	.171	-.401		.220	-1.598	-2.349	-2.082	-1.879	-1.413
.326	-.219	.369	-.049	-.103		.270	-1.355	-1.637	-1.656	-1.512	-1.071
.371	-.394	.465	-.462	-.586		.400	-.962	-1.184	-1.254	-1.196	-.994
.392	-.520	.530	-.936	-.790		.620	-.830	-1.165	-1.121	-1.056	-.987
.413	-.594	.599	-1.234	.204		.685	-2.560	-1.668	-.675	-1.603	-1.994
.434	-.627	.624	-1.955	.688		.693	-2.772	-1.487	-1.923	-1.904	-1.588
.457	-.542	.610	-1.474	.751		.700	-2.079	-1.138	-1.637	-1.354	-1.394
.481	-.478	.550	-1.253	.713		.720	-1.149	-.815	-.949	-.987	-.987
.502	-.471	.562	-1.094	.669		.750	-.812	-.637	-.904	-.981	-.994
.551	-.316	.540	-1.006	.639		.800	-.587	-.490	-.891	-.943	-.981
.585	-.265	.522	-1.139	.669		.900	-.462	-.471	-.841	-.905	-.936
.592	-.242	.478	-1.581	-1.490		.980	-.356	-.414	-.713	-.892	-.865
.613	-.161	.350	-1.303	-.950	Lower	.025	.812	.850	.853	.791	.697
.634	-.142	.261	-.753	-.560		.120	.928	.853	.750	.759	.613
.655	-.127	.127	-.450	-.344		.220	.880	.841	.828	.784	.671
.676	-.265	.013	-.228	-.383		.300	.793	.777	.771	.715	.620
.696	-.232	.045	-.114	.032		.620	.787	.821	.790	.715	.290
.774	-.013	.151	.019	.134		.750	.655	.872	.796	.671	.607
.852	-.116	.096	-.263	.032		.850	.693	.820	.611	.557	.497
.877	-.033	.146	-.025	.134		.950	.475	.401	.280	.207	.187

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TABLE 15(a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.5$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = -1.4°$											
.032	.248	.289	.284	.313	Upper	.010	.949	.849	.853	.828	.790
.053	.047	.083	.070	.060		.080	.499	.378	.313	.290	.330
.100	-.106	-.012	-.070	-.090		.130	-.274	-.478	-.601	-.568	-.590
.145	-.106	-.059	-.058	-.048		.145	-4.077	-3.764	-3.696	-4.171	-3.882
.189	-.035	-.024	-.012	.000		.155	-1.442	-1.451	-1.490	-1.471	-1.186
.234	-.071	.059	-.012	.012		.180	-1.083	-.908	-1.058	-.985	-1.085
.280	-.053	.071	-.029	.054		.220	-.596	-.631	-.745	-.840	-.714
.326	-.035	.083	-.035	.084		.270	-.467	-.549	-.655	-.597	-.572
.371	-.094	.136	-.075	.144		.400	-.432	-.425	-.649	-.695	-.596
.392	-.047	.139	.035	.355		.620	-.329	-.301	-.829	-.904	-.932
.413	-.136	.142	-.087	-.084		.685	-.986	-.661	-3.017	-6.041	-5.209
.434	-.165	.177	-.278	-.216		.693	-.925	-.920	-4.339	-6.516	-4.389
.457	-.201	.185	-.342	-.192		.700	-.742	-.844	-3.095	-3.968	-3.115
.480	-.248	.195	-.295	-.066		.720	-.596	-.637	-.913	-1.691	-1.144
.502	-.319	.210	-.278	.042		.750	-.463	-.649	-.601	-1.054	-.891
.551	-.336	.230	-.261	.276		.800	-.359	-.460	-.577	-.608	-.779
.585	-.342	.248	-.307	.234		.900	-.396	-.454	-.541	-.295	-.661
.592	-.342	.224	-.342	-.331		.980	-.359	-.419	-.445	-.029	-.513
.613	-.307	.159	-.301	-.285	Lower	.025	-.213	-.136	.030	.104	-.006
.634	-.271	.071	-.278	-.306		.120	-.341	-.100	.024	.075	-.041
.655	-.260	.006	-.249	-.403		.220	-.280	-.112	.000	.052	-.041
.675	-.201	-.077	-.203	-.445		.300	-.152	-.142	-.066	.000	-.077
.696	-.147	-.088	-.133	-.361		.620	.450	.242	.072	.006	-.189
.774	.041	.041	.046	.024		.750	.742	.448	.162	.122	.106
.852	-.047	-.041	-.006	-.102		.850	.694	.566	.343	.255	.271
.930	.024	-.130	.041	-.138		.950	.377	.354	.325	.330	.277
$\alpha = 5.7°$											
.032	.091	.468	.156	.306	Upper	.010	.630	.643	.556	.583	.548
.053	-.103	.219	-.036	.069		.080	-.115	-.137	-.219	-.276	-.225
.100	-.176	.069	-.180	-.119		.130	-1.235	-1.417	-1.573	-1.520	-1.613
.145	-.140	-.006	-.156	-.087		.145	-6.412	-6.006	-5.782	-6.364	-6.134
.189	-.061	.044	-.090	-.056		.155	-2.579	-2.579	-2.647	-2.632	-2.294
.234	-.091	.094	-.024	-.062		.180	-1.853	-1.667	-1.848	-1.749	-1.905
.280	-.091	.106	.024	-.037		.220	-1.127	-1.155	-1.311	-1.400	-1.321
.326	-.085	.119	.012	-.025		.270	-.828	-.943	-1.099	-1.076	-1.065
.371	-.164	.175	-.114	.000		.400	-.624	-.712	-.962	-1.064	-1.089
.392	-.210	.230	-.198	.175		.620	-.420	-.481	-1.086	-1.563	-1.649
.413	-.262	.275	-.463	.137		.685	-1.121	-.862	-3.565	-8.155	-10.127
.434	-.310	.293	-.709	.150		.693	-1.076	-1.099	-4.845	-8.738	-9.299
.457	-.323	.300	-.643	.287		.700	-.771	-.912	-3.490	-5.703	-7.090
.480	-.365	.320	-.565	.387		.720	-.579	-.631	-1.086	-2.626	-3.207
.502	-.432	.340	-.481	.393		.750	-.516	-.593	-.637	-1.689	-2.282
.551	-.420	.360	-.391	.462		.800	-.427	-.481	-.587	-1.094	-1.826
.585	-.414	.381	-.403	.487		.900	-.478	-.468	-.562	-.517	-1.308
.592	-.402	.375	-.421	-.425		.980	-.471	-.443	-.537	-.030	-.487
.613	-.365	.275	-.361	-.219	Lower	.025	.064	.318	.350	.385	.213
.634	-.335	.175	-.343	-.343		.120	.121	.268	.312	.288	.116
.655	-.310	.031	-.325	-.518		.220	.529	.300	.306	.276	.195
.675	-.237	-.119	-.288	-.475		.300	.592	.418	.487	.499	.505
.696	-.176	-.106	-.204	-.312		.620	.694	.668	.693	.655	.207
.774	.043	.062	.048	.025		.750	.821	.755	.737	.709	.578
.852	-.030	.019	.000	-.069		.850	.592	.593	.624	.631	.536
.930	.018	-.119	.060	-.094		.950	.312	.256	.325	.475	.341
$\alpha = 13.2°$											
.032	-.068	.637	-.069	.253	Upper	.010	-.164	-1.076	-1.309	-1.376	-.707
.053	-.205	.388	-.258	.044		.080	-.898	-.942	-1.050	-1.150	-.937
.100	-.155	.217	-.346	-.164		.130	-2.271	-2.490	-2.745	-2.721	-2.705
.145	-.112	.096	-.314	-.177		.145	-8.198	-8.074	-7.705	-8.610	-8.009
.189	-.068	.134	-.258	-.139		.155	-3.429	-3.668	-3.808	-3.928	-3.275
.234	-.093	.204	-.069	-.145		.180	-2.315	-2.343	-2.594	-2.577	-2.562
.280	-.112	.217	.057	-.164		.220	-1.442	-1.630	-1.822	-1.961	-1.756
.326	-.112	.229	.031	-.190		.270	-1.044	-1.286	-1.468	-1.496	-1.408
.371	-.230	.312	-.201	-.202		.400	-.784	-.866	-1.132	-1.307	-1.359
.392	-.285	.350	-.547	-.152		.620	-.576	-.624	-1.101	-2.036	-1.998
.413	-.341	.427	-.892	.196		.685	-1.202	-.853	-2.802	-8.402	-11.867
.434	-.391	.458	-1.156	.506		.693	-1.177	-1.019	-3.928	-9.207	-11.017
.457	-.385	.460	-.930	.595		.700	-.810	-.860	-2.701	-5.933	-8.393
.480	-.409	.465	-.773	.557		.720	-.582	-.605	-.784	-2.790	-3.815
.502	-.465	.470	-.672	.487		.750	-.493	-.516	-.614	-1.785	-2.692
.551	-.403	.470	-.553	.544		.800	-.405	-.388	-.531	-1.119	-2.103
.585	-.378	.471	-.547	.620		.900	-.430	-.350	-.399	-.522	-1.632
.592	-.366	.446	-.578	-.493		.980	-.493	-.350	-.449	-.057	-.900
.613	-.304	.337	-.465	-.354	Lower	.025	.455	.662	.658	.685	.533
.634	-.248	.217	-.396	-.386		.120	.797	.802	.753	.723	.540
.655	-.236	.070	-.358	-.481		.220	.753	.777	.746	.735	.583
.675	-.192	-.064	-.321	-.405		.300	.645	.688	.677	.635	.502
.696	-.149	-.051	-.245	-.247		.620	.734	.809	.765	.679	.074
.774	.037	.159	.025	.076		.750	.791	.879	.772	.704	.552
.852	-.050	.083	-.050	-.070		.850	.588	.650	.620	.628	.447
.930	.012	.038	.000	.025		.950	.310	.325	.316	.459	.223

TABLE 15 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = .0 h_d/c = 0.5

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					α = 18.9°						
.032	-.166	.753	-.316	.132	Upper	.010	-3.783	-2.9[illegible]5	-3.227	-3.291	-3.103
.053	-.312	.524	-.426	-.053		.080	-1.638	-1.9[illegible]7	-2.687	-2.956	-1.883
.100	-.186	.309	-.529	-.290		.130	-2.973	-3.0[illegible]9	-3.188	-3.162	-3.534
.145	-.172	.202	-.478	-.316		.145	-9.261	-8.9[illegible]0	-8.437	-8.970	-9.549
.189	-.093	.229	-.426	-.250		.155	-3.973	-4.4[illegible]1	-4.511	-4.537	-4.251
.234	-.086	.276	-.155	-.303		.180	-2.606	-2.9[illegible]5	-3.135	-3.104	-3.369
.280	-.119	.296	.110	-.329		.220	-1.581	-2.0[illegible]7	-2.239	-2.343	-2.347
.326	-.159	.316	.045	-.395		.270	-1.151	-1.5[illegible]7	-1.791	-1.833	-1.936
.371	-.371	.410	-.271	-.435		.400	-.810	-1.0[illegible]6	-1.297	-1.497	-1.844
.392	-.420	.460	-.826	-.533		.620	-.550	-.7[illegible]3	-1.106	-2.130	-2.241
.413	-.484	.524	-1.058	.178		.685	-1.328	-.8[illegible]4	-1.423	-6.866	-11.698
.434	-.544	.565	-1.600	.593		.693	-1.284	-1.0[illegible]6	-2.509	-7.447	-10.769
.457	-.491	.570	-1.271	.672		.700	-.886	-.8[illegible]4	-1.745	-4.763	-8.156
.480	-.484	.550	-.955	.639		.720	-.601	-.5[illegible]5	-.685	-2.233	-3.667
.502	-.504	.530	-.826	.593		.750	-.481	-.4[illegible]8	-.593	-1.413	-2.599
.551	-.378	.510	-.684	.580		.800	-.392	-.4[illegible]4	-.514	-.871	-2.082
.585	-.358	.491	-.665	.645		.900	-.399	-.3[illegible]7	-.421	-.484	-1.664
.592	-.338	.457	-.710	-.652		.980	-.405	-.3[illegible]6	-.448	-.213	-1.074
.613	-.252	.370	-.678	-.257							
.634	-.212	.242	-.613	-.408	Lower	.025	.690	.820	.790	.800	.650
.655	-.206	.081	-.516	-.468		.120	.873	.847	.764	.729	.550
.675	-.119	-.040	-.407	-.375		.220	.816	.807	.797	.774	.617
.696	-.073	-.007	-.297	-.263		.300	.696	.740	.692	.691	.531
.774	.053	.175	.013	-.170		.620	.759	.8[illegible]4	.764	.697	.086
.852	-.066	.067	-.039	-.092		.750	.829	.8[illegible]1	.803	.703	.564
.930	-.007	.108	-.045	.138		.850	.639	.6[illegible]6	.619	.645	.477
						.950	.405	.3[illegible]3	.329	.439	.225
					α = 23.0°						
.032	-.226	.842	-.420	.013	Upper	.010	-7.553	-3.8[illegible]0	-3.996	-3.649	-3.491
.053	-.368	.610	-.522	-.156		.080	-1.961	-3.5[illegible]1	-4.080	-3.706	-2.136
.100	-.213	.398	-.592	-.338		.130	-3.214	-2.9[illegible]1	-3.157	-3.006	-2.426
.145	-.174	.292	-.541	-.403		.145	-9.464	-8.0[illegible]4	-7.017	-6.253	-6.395
.189	-.103	.332	-.478	-.338		.155	-4.036	-4.3[illegible]0	-4.048	-3.407	-2.678
.234	-.071	.371	-.146	-.403		.180	-2.619	-2.9[illegible]1	-2.930	-2.426	-2.136
.280	-.129	.378	.146	-.403		.220	-1.581	-2.0[illegible]9	-2.079	-1.777	-1.323
.326	-.207	.385	.013	-.533		.270	-1.765	-1.6[illegible]1	-1.644	-1.369	-.987
.371	-.387	.497	-.414	-.611		.400	-.898	-1.1[illegible]4	-1.182	-1.032	-.962
.392	-.485	.560	-.936	-.793		.620	-.677	-.948	-1.065	-.974	-.897
.413	-.581	.610	-1.178	.156		.685	-1.891	-1.2[illegible]3	-.741	-1.637	-1.891
.434	-.600	.623	-1.942	.624		.693	-1.822	-1.3[illegible]9	-1.910	-1.834	-1.549
.457	-.561	.630	-1.414	.689		.700	-1.290	-1.1[illegible]4	-1.540	-1.261	-1.355
.480	-.497	.610	-1.153	.650		.720	-.822	-.7[illegible]3	-.858	-.898	-.923
.502	-.497	.590	-.987	.585		.750	-.671	-.6[illegible]0	-.806	-.885	-.923
.551	-.336	.550	-.841	.591		.800	-.563	-.6[illegible]3	-.715	-.828	-.916
.585	-.297	.537	-.866	.637		.900	-.588	-.5[illegible]4	-.689	-.853	-.884
.592	-.271	.504	-1.089	-.988		.980	-.506	-.4[illegible]8	-.559	-.815	-.852
.613	-.207	.398	-1.095	-.780							
.634	-.200	.259	-.796	-.591	Lower	.025	.784	.8[illegible]5	.884	.860	.691
.655	-.168	.106	-.522	-.513		.120	.886	.8[illegible]5	.793	.790	.613
.675	-.110	-.013	-.318	-.286		.220	.860	.8[illegible]5	.832	.821	.678
.696	-.071	.027	-.197	-.097		.300	.759	.7[illegible]2	.760	.771	.620
.774	-.019	.212	.019	.117		.620	.810	.8[illegible]9	.786	.764	.290
.852	-.116	.119	-.051	-.019		.750	.886	.9[illegible]2	.845	.751	.613
.930	-.058	.159	-.025	.169		.850	.671	.7[illegible]3	.624	.618	.516
						.950	.418	.3[illegible]8	.286	.274	.161

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TABLE 15 Continued (b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = -1.4°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.268	.289	.287	.293
.053	.091	.083	.108	.078
.100	-.073	-.029	-.060	-.078
.145	-.061	-.059	-.018	-.042
.189	.012	-.012	.030	[illegible]
.234	-.018	.047	.006	.024
.280	-.024	.065	-.017	.024
.326	.024	.083	-.017	.078
.371	-.061	.124	.125	.149
.392	-.030	.130	.096	.334
.413	-.079	.142	-.054	-.102
.434	-.128	.153	-.227	-.257
.457	-.146	.160	-.287	-.203
.480	-.195	.170	-.233	-.096
.502	-.268	.185	-.197	.030
.551	-.304	.200	-.137	.245
.585	-.316	.206	-.221	.221
.592	-.323	.206	-.352	-.078
.613	-.286	.153	-.317	-.006
.634	-.268	.094	-.305	-.328
.655	-.256	-.006	-.311	-.448
.675	-.195	-.100	-.269	-.496
.696	-.152	-.100	-.203	-.424
.774	.037	.047	.030	-.036
.852	-.037	-.047	.000	-.090
.930	.037	-.130	.000	-.113

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.955	.849	.848	.842	.840
	.080	.514	.413	.346	.317	.359
	.130	-.230	-.413	-.543	-.526	-.602
	.145	-3.882	-3.598	-3.458	-4.097	-3.834
	.155	-1.367	-1.321	-1.380	-1.427	-1.193
	.180	-.992	-.820	-.973	-.926	-1.071
	.220	-.538	-.555	-.669	-.818	-.706
	.270	-.387	-.437	-.543	-.555	-.548
	.400	-.296	-.319	-.520	-.651	-.560
	.620	-.030	-.035	-.651	-.758	-.901
	.685	-.883	-.578	-2.478	-6.086	-5.428
	.693	-.835	-.826	-3.601	-6.719	-4.479
	.700	-.689	-.743	-2.502	-4.049	-3.195
	.720	-.653	-.543	-.699	-1.714	-1.132
	.750	-.466	-.619	-.615	-1.033	-.858
	.800	-.351	-.525	-.567	-.555	-.749
	.900	-.387	-.472	-.543	-.227	-.639
	.980	-.339	-.460	-.460	.030	-.499
Lower	.025	-.224	-.153	.024	.078	.006
	.120	-.302	-.130	.018	.042	-.043
	.220	-.260	-.147	-.024	.024	-.043
	.300	-.139	-.177	-.060	-.006	-.067
	.620	.496	.248	.054	.012	-.219
	.750	.786	.460	.149	.119	.097
	.850	.695	.560	.293	.239	.268
	.950	.357	.348	.269	.346	.292

$\alpha = 5.8°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.112	.474	.150	.306
.053	-.087	.240	-.044	.070
.100	-.167	.057	-.175	-.121
.145	-.118	-.019	-.144	-.108
.189	-.037	.051	-.087	-.038
.234	-.074	.108	-.037	-.038
.280	-.074	.114	.044	-.032
.326	-.062	.133	.006	-.013
.371	-.136	.202	-.100	.051
.392	-.190	.236	-.169	.210
.413	-.236	.266	-.462	.121
.434	-.285	.297	-.699	.166
.457	-.292	.300	-.612	.267
.480	-.341	.320	-.500	.376
.502	-.397	.340	-.406	.382
.551	-.397	.360	-.287	.446
.585	-.391	.367	-.331	.497
.592	-.409	.367	-.475	-.420
.613	-.354	.253	-.412	-.223
.634	-.335	.145	-.387	-.420
.655	-.310	.006	-.437	-.573
.675	-.236	-.152	-.387	-.573
.696	-.192	-.177	-.312	-.452
.774	.012	.038	-.025	-.076
.852	-.050	.038	-.012	-.102
.930	.000	-.114	-.006	-.108

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.688	.658	.592	.556	.596
	.080	-.032	-.101	-.197	-.275	-.174
	.130	-1.127	-1.303	-1.541	-1.523	-1.476
	.145	-6.132	-5.757	-5.636	-6.281	-5.887
	.155	-2.426	-2.442	-2.566	-2.622	-2.177
	.180	-1.776	-1.544	-1.758	-1.723	-1.793
	.220	-1.025	-1.056	-1.216	-1.374	-1.241
	.270	-.739	-.848	-1.006	-1.024	-1.024
	.400	-.478	-.544	-.853	-.999	-1.005
	.620	-.197	-.234	-.815	-1.523	-1.538
	.685	-1.038	-.867	-2.413	-7.655	-9.864
	.693	-1.006	-.962	-3.470	-8.404	-9.045
	.700	-.796	-.867	-2.369	-5.376	-6.861
	.720	-.630	-.620	-.739	-2.423	-3.040
	.750	-.579	-.582	-.700	-1.542	-2.146
	.800	-.465	-.443	-.669	-.962	-1.650
	.900	-.503	-.474	-.554	-.456	-1.179
	.980	-.503	-.424	-.592	-.025	-.434
Lower	.025	.032	.291	.306	.356	.199
	.120	.121	.266	.280	.268	.118
	.220	.427	.291	.261	.262	.112
	.300	.567	.418	.458	.462	.459
	.620	.732	.683	.688	.656	.223
	.750	.828	.772	.751	.724	.602
	.850	.592	.595	.599	.618	.546
	.950	.280	.278	.261	.462	.372

$\alpha = 13.3°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.055	.630	-.098	.244
.053	-.225	.396	-.262	.040
.100	-.184	.201	-.379	-.178
.145	-.143	.123	-.307	-.171
.189	-.075	.149	-.262	-.138
.234	-.102	.201	-.118	-.118
.280	-.102	.208	.072	-.178
.326	-.116	.214	.000	-.224
.371	-.245	.305	-.209	-.198
.392	.075	.355	-.536	-.145
.413	-.321	.403	-.850	.198
.434	-.389	.448	-1.119	.520
.457	-.396	.450	-.909	.593
.480	-.396	.440	-.726	.540
.502	-.484	.425	-.589	.501
.551	-.430	.410	-.406	.520
.585	-.409	.409	-.419	.599
.592	-.402	.416	-.549	-.474
.613	-.321	.312	-.491	-.362
.634	-.307	.169	-.471	-.487
.655	-.307	.032	-.477	-.580
.675	-.211	-.117	-.432	-.540
.696	-.170	-.123	-.347	-.408
.774	.048	.097	-.026	-.260
.852	-.020	.039	-.065	-.132
.930	.027	.019	-.039	.026

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.000	-.682	-1.264	-1.256	-.634
	.080	-.779	-.858	-1.067	-1.145	-1.016
	.130	-2.105	-2.378	-2.740	-2.695	-2.878
	.145	-7.868	-7.822	-7.699	-8.641	-8.483
	.155	-3.205	-3.502	-3.747	-3.898	-3.471
	.180	-2.137	-2.202	-2.529	-2.538	-2.707
	.220	-1.269	-1.501	-1.758	-1.943	-1.855
	.270	-.867	-1.137	-1.383	-1.439	-1.493
	.400	-.591	-.669	-1.021	-1.269	-1.405
	.620	-.383	-.364	-.915	-2.002	-2.073
	.685	-1.100	-.858	-1.482	-8.203	-12.173
	.693	-1.018	-.760	-2.476	-8.987	-11.232
	.700	-.742	-.734	-1.640	-5.776	-8.531
	.720	-.515	-.559	-.685	-2.630	-3.798
	.750	-.452	-.585	-.626	-1.688	-2.666
	.800	-.446	-.364	-.566	-1.066	-2.080
	.900	-.465	-.396	-.501	-.543	-1.657
	.980	-.578	-.461	-.547	-.203	-1.002
Lower	.025	.440	.669	.659	.648	.559
	.120	.779	.825	.764	.720	.586
	.220	.779	.760	.764	.720	.627
	.300	.660	.682	.678	.654	.525
	.620	.767	.825	.784	.693	.116
	.750	.823	.890	.784	.733	.607
	.850	.591	.682	.593	.641	.491
	.950	.302	.305	.263	.451	.252

TABLE 15 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	Fuselage: 0.000, Upper surface	Fuselage: 0.000, Lower surface	Fuselage: 0.154, Upper surface	Fuselage: 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
							C_p values for spanwise stations, $\frac{y}{b/2}$, of:				
$\alpha = 18.9°$											
.032	-.183	.758	-.287	.167	Upper	.010	-3.513	-2.471	-2.923	-2.987	-2.760
.053	-.314	.548	-.420	-.026		.080	-1.538	-1.420	-2.410	-2.573	-1.603
.100	-.209	.331	-.484	-.256		.130	-2.878	-2.723	-2.910	-2.942	-3.297
.145	-.164	.229	-.446	-.295		.145	-8.987	-8.164	-7.750	-8.495	-8.929
.189	-.078	.248	-.395	-.250		.155	-3.801	-3.910	-4.051	-4.196	-3.931
.234	-.072	.312	-.172	-.295		.180	-2.487	-2.496	-2.801	-2.834	-3.048
.280	-.111	.318	.076	-.301		.220	-1.462	-1.700	-1.936	-2.101	-2.119
.326	-.144	.325	.238	-.172		.270	-1.026	-1.274	-1.494	-1.617	-1.727
.371	-.314	.427	-.248	-.410		.400	-.641	-.713	-1.013	-1.274	-1.616
.392	-.375	.470	-.771	-.481		.620	-.372	-.337	-.756	-1.955	-1.982
.413	-.445	.509	-1.006	.179		.685	-1.090	-.637	-.301	-5.801	-10.701
.434	-.491	.573	-1.516	.596		.693	-1.051	-.796	-1.462	-6.317	-9.818
.457	-.458	.560	-1.177	.667		.700	-.763	-.713	-1.128	-3.903	-7.405
.480	-.445	.540	-.866	.622		.720	-.538	-.535	-.571	-1.681	-3.205
.502	-.464	.520	-.688	.590		.750	-.487	-.509	-.545	-1.019	-2.224
.551	-.373	.500	-.465	.564		.800	-.449	-.337	-.564	-.650	-1.766
.585	-.353	.509	-.503	.635		.900	-.455	-.357	-.449	-.490	-1.393
.592	-.340	.484	-.573	-.526		.980	-.526	-.357	-.500	-.350	-.863
.613	-.281	.357	-.497	-.321	Lower	.025	.686	.82	.808	.809	.667
.634	-.268	.217	-.458	-.474		.120	.885	.86	.756	.713	.563
.655	-.275	.064	-.446	-.538		.220	.827	.81	.808	.783	.608
.675	-.177	-.064	-.401	-.481		.300	.731	.75	.699	.713	.563
.696	-.150	-.045	-.331	-.385		.620	.814	.84	.782	.713	.118
.774	.007	.166	-.045	-.275		.750	.872	.90	.776	.739	.595
.852	-.072	.076	-.070	-.173		.850	.667	.71	.609	.637	.497
.930	-.020	.159	-.115	.141		.950	.359	.37	.288	.414	.288
$\alpha = 23.1°$											
.032	-.212	.808	-.432	.026	Upper	.010	-7.228	-3.56	-3.702	-3.446	-3.199
.053	-.321	.614	-.549	-.124		.080	-1.820	-3.245	-3.794	-3.446	-1.724
.100	-.192	.414	-.587	-.360		.130	-3.001	-2.79	-2.917	-2.833	-2.333
.145	-.160	.294	-.536	-.406		.145	-8.938	-7.79	-6.822	-6.324	-6.231
.189	-.077	.321	-.471	-.340		.155	-3.743	-4.06	-3.807	-3.311	-2.513
.234	-.051	.381	-.187	-.392		.180	-2.426	-2.72	-2.701	-2.323	-1.974
.280	-.103	.387	.123	-.406		.220	-1.433	-1.89	-1.871	-1.639	-1.205
.326	-.192	.407	.110	-.510		.270	-1.129	-1.40	-1.439	-1.245	-.891
.371	-.372	.494	-.413	-.549		.400	-.749	-.85	-.948	-.891	-.840
.392	.115	.554	-.884	-.739		.620	-.407	-.45	-.759	-.787	-.808
.413	-.551	.614	-1.129	.196		.685	-1.349	-.90	-.406	-1.800	-1.679
.434	-.564	.641	-1.833	.667		.693	-1.278	-1.01	-1.727	-2.110	-1.442
.457	-.519	.620	-1.271	.726		.700	-.936	-1.00	-1.361	-1.362	-1.276
.480	-.449	.590	-1.013	.674		.720	-.703	-.80	-.680	-.903	-.821
.502	-.442	.575	-.832	.634		.750	-.626	-.65	-.667	-.884	-.840
.551	-.308	.555	-.613	.602		.800	-.574	-.53	-.615	-.826	-.840
.585	-.263	.534	-.658	.641		.900	-.568	-.58	-.523	-.832	-.846
.592	-.256	.494	-.781	-.726		.980	-.587	-.56	-.510	-.768	-.769
.613	-.173	.381	-.665	-.660	Lower	.025	.794	.88	.863	.865	.718
.634	-.179	.260	-.652	-.589		.120	.910	.88	.805	.800	.635
.655	-.154	.060	-.652	-.641		.220	.865	.88	.850	.820	.686
.675	-.083	-.080	-.568	-.530		.300	.774	.77	.746	.761	.628
.696	-.071	-.033	-.419	-.334		.620	.826	.88	.811	.742	.327
.774	[illegible]	.200	-.032	-.230		.750	.878	.92	.837	.781	.641
.852	-.090	.093	-.077	-.131		.850	.678	.73	.641	.639	.526
.930	-.077	.180	-.123	.196		.950	.394	.34	.288	.284	.218

L-927

L-9726

TABLE 15 Continued
(a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 4.0 h_d/c = 2.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = -1.3°

x/l	Fuselage 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.276	.294	.285	.296
.053	.082	.073	.083	.062
.100	-.041	-.031	-.095	-.099
.145	-.047	-.067	-.053	-.043
.189	.012	-.024	[illegible]	[illegible]
.234	-.012	.037	-.024	[illegible]
.280	-.006	.067	-.059	.018
.326	.012	.067	-.030	.062
.371	-.029	.110	-.059	.160
.392	-.035	.119	.083	.370
.413	-.059	.129	-.036	-.185
.434	-.106	.153	-.214	-.333
.457	-.117	.160	-.226	-.271
.480	-.164	.170	-.172	-.160
.502	-.217	.180	-.107	-.031
.551	-.270	.190	-.024	.222
.585	-.305	.202	-.190	.222
.592	-.340	.214	-.415	-.413
.613	-.328	.147	-.392	-.086
.634	-.317	.061	-.439	-.345
.655	-.328	-.012	-.457	-.481
.675	-.270	-.135	-.421	-.567
.696	-.211	-.171	-.344	-.586
.774	.023	-.037	-.095	-.160
.852	-.016	-.049	-.006	-.129
.930	.047	-.153	.006	-.160

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.962	.882	.838	.831	.815
	.080	.553	.465	.382	.380	.399
	.130	-.138	-.331	-.487	-.463	-.457
	.145	-3.570	-3.387	-3.347	-3.805	-3.477
	.155	-1.184	-1.213	-1.294	-1.282	-1.008
	.180	-.841	-.710	-.900	-.819	-.926
	.220	-.421	-.459	-.592	-.724	-.604
	.270	-.264	-.343	-.481	-.499	-.463
	.400	-.120	-.153	-.431	-.564	-.469
	.620	.294	.355	-.518	-.789	-.786
	.685	-.811	-.447	-1.960	-5.757	-5.230
	.693	-.835	-.753	-3.082	-6.179	-4.315
	.700	-.703	-.674	-2.108	-3.722	-3.002
	.720	-.655	-.508	-.690	-1.496	-.938
	.750	-.613	-.637	-.641	-.926	-.686
	.800	-.367	-.576	-.586	-.528	-.610
	.900	-.361	-.429	-.604	-.237	-.575
	.980	-.325	-.447	-.487	-.024	-.440
Lower	.025	-.264	-.251	-.031	.036	.023
	.120	-.349	-.196	-.043	.024	-.018
	.220	-.319	-.202	-.062	.006	-.023
	.300	-.192	-.257	-.117	-.012	-.047
	.620	.511	.276	-.043	-.107	-.276
	.750	.817	.490	.037	-.006	-.012
	.850	.673	.588	.271	.131	.170
	.950	.373	.355	.277	.261	.252

α = 5.9°

x/l	Fuselage 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.091	.481	.170	.333
.053	-.079	.228	-.037	.099
.100	-.152	.089	-.170	-.105
.145	-.122	.013	-.134	-.080
.189	-.055	.063	-.061	-.037
.234	-.085	.120	-.024	-.031
.280	-.085	.120	.024	-.018
.326	-.067	.152	.024	-.012
.371	-.140	.202	-.073	.049
.392	-.175	.234	-.152	.222
.413	-.207	.266	-.365	.099
.434	-.250	.304	-.572	.099
.457	-.256	.310	-.499	.234
.480	-.298	.320	-.359	.339
.502	-.353	.335	-.243	.370
.551	-.353	.350	-.110	.444
.585	-.389	.367	-.243	.512
.592	-.426	.367	-.426	-.438
.613	-.426	.278	-.432	-.222
.634	-.414	.171	-.475	-.394
.655	-.444	.006	-.475	-.524
.675	-.389	-.145	-.456	-.592
.696	-.377	-.183	-.426	-.536
.774	-.037	-.013	-.250	-.210
.852	-.018	.025	-.018	-.105
.930	.006	-.114	[illegible]	-.117

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.726	.727	.660	.663	.609
	.080	.006	[illegible]	-.068	-.146	-.146
	.130	-.999	-1.101	-1.245	-1.266	-1.412
	.145	-5.602	-5.187	-4.882	-5.684	-5.574
	.155	-2.177	-2.138	-2.145	-2.246	-2.057
	.180	-1.557	-1.290	-1.455	-1.479	-1.680
	.220	-.887	-.835	-.999	-1.181	-1.181
	.270	-.565	-.607	-.789	-.858	-.931
	.400	-.279	-.240	-.598	-.822	-.937
	.620	.192	.266	-.542	-1.369	-1.454
	.685	-.912	-.455	-1.596	-7.017	-9.384
	.693	-.912	-.715	-2.650	-7.680	-8.617
	.700	-.726	-.696	-1.818	-4.856	-6.524
	.720	-.602	-.531	-.666	-2.124	-2.793
	.750	-.620	-.588	-.604	-1.327	-1.917
	.800	-.459	-.601	-.549	-.822	-1.467
	.900	-.428	-.468	-.623	-.402	-1.059
	.980	-.447	-.455	-.536	-.037	-.371
Lower	.025	.006	.272	.308	.341	.189
	.120	.056	.253	.308	.256	.097
	.220	.416	.247	.240	.237	.085
	.300	.565	.335	.314	.353	.426
	.620	.738	.721	.653	.657	.256
	.750	.862	.810	.740	.706	.609
	.850	.602	.639	.579	.645	.566
	.950	.298	.278	.271	.469	.365

α = 13.4°

x/l	Fuselage 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.057	.624	-.089	.220
.053	-.240	.406	-.247	.025
.100	-.190	.225	-.342	-.176
.145	-.139	.131	-.304	-.195
.189	-.089	.156	-.253	-.132
.234	-.095	.200	-.108	-.151
.280	-.101	.225	.057	-.151
.326	-.101	.225	.025	-.189
.371	-.226	.300	-.164	-.176
.392	-.260	.360	-.481	-.126
.413	-.285	.418	-.765	.176
.434	-.335	.450	-.999	.490
.457	-.342	.445	-.765	.553
.480	-.348	.440	-.550	.528
.502	-.418	.435	-.392	.465
.551	-.367	.430	-.202	.490
.585	-.373	.425	-.316	.578
.592	-.411	.418	-.430	-.333
.613	-.348	.300	-.519	-.365
.634	-.392	.175	-.531	-.459
.655	-.424	.031	-.531	-.553
.675	-.342	-.112	-.519	-.566
.696	-.323	-.150	-.468	-.553
.774	-.133	.062	-.215	-.335
.852	-.032	.075	-.076	-.182
.930	-.006	-.025	-.051	.025

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.063	-.062	-.654	-.981	-.267
	.080	-.698	-.674	-.842	-.987	-.822
	.130	-1.936	-2.042	-2.357	-2.404	-2.524
	.145	-7.347	-6.906	-6.800	-7.831	-7.604
	.155	-2.941	-3.016	-3.243	-3.473	-3.062
	.180	-1.917	-1.842	-2.137	-2.214	-2.379
	.220	-1.087	-1.193	-1.452	-1.670	-1.613
	.270	-.723	-.868	-1.094	-1.240	-1.265
	.400	-.408	-.362	-.742	-1.056	-1.189
	.620	-.069	[illegible]	-.729	-1.835	-1.784
	.685	-.980	-.762	-1.357	-7.591	-11.115
	.693	-.798	-.537	-2.332	-8.072	-10.286
	.700	-.622	-.612	-1.697	-5.035	-7.806
	.720	-.503	-.481	-.666	-2.239	-3.454
	.750	-.490	-.524	-.584	-1.392	-2.391
	.800	-.496	-.537	-.528	-.879	-1.835
	.900	-.465	-.437	-.578	-.436	-1.404
	.980	-.522	-.456	-.547	-.240	-.709
Lower	.025	.446	.612	.635	.671	.538
	.120	.767	.812	.742	.721	.582
	.220	.779	.737	.742	.715	.607
	.300	.660	.668	.654	.656	.538
	.620	.786	.849	.767	.683	.139
	.750	.848	.899	.804	.715	.595
	.850	.603	.562	.603	.645	.500
	.950	.333	.306	.264	.468	.266

TABLE 15 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 2.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	).426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing , flap , or aileron				
					α = 19.1°						
.032	-.158	.692	-.299	.154	Upper	.010	-3.041	-2.256	-2.603	-2.742	-2.542
.053	-.290	.468	-.455	-.051		.080	-1.429	-1.192	-1.840	-2.222	-1.594
.100	-.184	.288	-.507	-.269		.130	-2.729	-2.628	-2.795	-2.826	-3.181
.145	-.171	.205	-.455	-.295		.145	-8.556	-7.808	-7.417	-8.264	-8.614
.189	-.079	.218	-.403	-.237		.155	-3.580	-3.667	-3.776	-3.989	-3.793
.234	-.066	.244	-.175	-.276		.180	-2.293	-2.308	-2.564	-2.651	-2.898
.280	-.092	.282	.058	-.308		.220	-1.325	-1.526	-1.724	-1.962	-2.009
.326	-.119	.308	.013	-.391		.270	-.890	-1.058	-1.295	-1.468	-1.620
.371	-.303	.397	-.240	-.372		.400	-.494	-.436	-.756	-1.117	-1.488
.392	-.350	.460	-.708	-.442		.620	-.104	-.038	-.583	-1.949	-1.923
.413	-.395	.513	-.949	.186		.685	-.845	-.410	-.167	-5.185	-10.788
.434	-.461	.551	-1.410	.590		.693	-.845	-.603	-1.263	-5.607	-9.938
.457	-.435	.540	-1.065	.654		.700	-.637	-.647	-1.026	-3.365	-7.482
.480	-.408	.525	-.702	.622		.720	-.533	-.500	-.558	-1.377	-3.194
.502	-.441	.510	-.520	.571		.750	-.533	-.506	-.532	-.819	-2.147
.551	-.342	.490	-.240	.526		.800	-.539	-.571	-.532	-.598	-1.587
.585	-.356	.487	-.318	.622		.900	-.487	-.442	-.603	-.585	-1.133
.592	-.382	.462	-.442	-.404		.980	-.487	-.449	-.545	-.455	-.547
.613	-.303	.321	-.507	-.365							
.634	-.356	.212	-.507	-.474	Lower	.025	.656	.788	.782	.793	.672
.655	-.356	.032	-.526	-.564		.120	.871	.846	.769	.741	.580
.675	-.277	-.103	-.500	-.583		.220	.812	.795	.795	.780	.606
.696	-.244	-.103	-.442	-.571		.300	.728	.724	.699	.689	.580
.774	-.145	.090	-.247	-.410		.620	.793	.865	.795	.702	.165
.852	-.040	.058	-.097	-.250		.750	.884	.910	.769	.728	.619
.930	-.007	.154	-.130	.141		.850	.630	.692	.622	.630	.527
						.950	.377	.333	.295	.370	.323
					α = 23.1°						
.032	-.216	.790	-.452	-.006	Upper	.010	-5.822	3.220	-3.273	-3.246	-3.026
.053	-.314	.573	-.536	-.146		.080	-1.844	2.832	-3.286	-3.169	-1.622
.100	-.179	.382	-.607	-.363		.130	-2.845	2.562	-2.624	-2.710	-2.295
.145	-.128	.310	-.561	-.414		.145	-8.456	7.258	-6.444	-6.479	-6.147
.189	-.064	.316	-.471	-.344		.155	-3.510	3.688	-3.470	-3.317	-2.449
.234	-.032	.342	-.200	-.395		.180	-2.213	2.410	-2.401	-2.278	-1.910
.280	-.071	.369	.071	-.382		.220	-1.317	1.633	-1.611	-1.594	-1.167
.326	-.160	.395	-.071	-.478		.270	-1.041	1.139	-1.172	-1.155	-.833
.371	-.359	.487	-.394	-.560		.400	-.612	-.527	-.669	-.755	-.833
.392	-.440	.540	-.800	-.675		.620	-.184	-.132	-.573	-.884	-.763
.413	-.526	.593	-1.052	.153		.685	-.988	-.501	-.051	-2.188	-1.821
.434	-.545	.606	-1.749	.611		.693	-.988	-.685	-1.216	-2.355	-1.603
.457	-.513	.595	-1.194	.637		.700	-.803	-.797	-1.025	-1.478	-1.397
.480	-.442	.575	-.929	.611		.720	-.659	-.685	-.637	-.807	-.833
.502	-.429	.550	-.665	.554		.750	-.672	-.817	-.643	-.820	-.833
.551	-.308	.530	-.400	.541		.800	-.678	-.639	-.624	-.813	-.846
.585	-.282	.514	-.458	.592		.900	-.645	-.547	-.503	-.781	-.846
.592	-.321	.474	-.652	-.541		.980	-.652	-.547	-.497	-.652	-.808
.613	-.244	.316	-.645	-.509							
.634	-.269	.204	-.632	-.630	Lower	.025	.790	.876	.853	.832	.712
.655	-.276	.013	-.626	-.694		.120	.922	.869	.796	.807	.635
.675	-.192	-.138	-.568	-.681		.220	.856	.843	.815	.807	.692
.696	-.141	-.112	-.484	-.579		.300	.771	.751	.732	.749	.641
.774	-.103	-.040	-.071	-.400		.620	.830	.889	.796	.742	.321
.852	-.077	.040	-.071	-.217		.750	.896	.909	.809	.691	.647
.930	-.096	.184	-.219	.153		.850	.659	.724	.611	.587	.532
						.950	.362	.336	.267	.265	.231

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TABLE 15 Continued

(d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.0$ $h_d/c = 3.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = -1.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
.032	.257	.306	.287	.304	Upper	.010	.962	.677	.640	.833	.800
.053	.066	.084	.088	.073		.080	.571	.511	.414	.387	.388
.100	-.084	-.024	-.064	-.067		.130	-.096	-.228	-.371	-.393	-.454
.145	-.078	-.072	-.041	-.030		.145	-3.407	-3.041	-2.958	-3.553	-3.446
.189	-.006	-.006	.023	.000		.155	-1.100	-1.010	-1.059	-1.132	-.962
.234	-.030	.102	-.006	.024		.180	-.763	-.553	-.730	-.721	-.926
.280	-.030	.072	-.041	.061		.220	-.355	-.300	-.450	-.645	-.603
.326	-.006	.096	-.035	.116		.270	-.198	-.198	-.335	-.422	-.454
.371	-.042	.126	-.023	.219		.400	.018	.090	-.268	-.487	-.472
.392	-.060	.126	.117	.353		.620	.637	.685	-.359	-.750	-.800
.413	-.054	.126	.018	-.189		.685	-.775	-.349	-1.217	-5.541	-5.596
.434	-.090	.144	-.135	-.329		.693	-.793	-.691	-2.264	-6.162	-4.724
.457	-.108	.140	-.158	-.286		.700	-.649	-.643	-1.637	-3.641	-3.237
.480	-.137	.160	-.070	-.158		.720	-.547	-.469	-.615	-1.431	-.991
.502	-.221	.170	.059	-.037		.750	-.571	-.475	-.554	-.850	-.693
.551	-.269	.160	.094	.231		.800	-.439	-.541	-.511	-.446	-.615
.585	-.340	.192	-.205	.256		.900	-.409	-.439	-.548	-.211	-.555
.592	-.400	.192	-.440	-.414		.980	-.409	-.445	-.487	.023	-.418
.613	-.472	.156	-.451	-.134	Lower	.025	-.306	-.228	-.049	.018	-.006
.634	-.418	.066	-.463	-.359		.120	-.349	-.198	-.067	.012	-.030
.655	-.436	-.024	-.457	-.444		.220	-.343	-.216	-.085	-.012	-.042
.675	-.382	-.150	-.440	-.523		.300	-.216	-.252	-.122	-.053	-.072
.696	-.364	-.192	-.405	-.499		.620	.535	.300	-.049	-.117	-.263
.774	-.090	-.024	-.229	-.231		.750	.835	.499	.037	-.012	-.042
.852	.006	-.054	.018	-.103		.850	.679	.595	.256	.106	.125
.930	.018	-.126	.023	-.134		.950	.343	.349	.268	.258	.245

$\alpha = 6.0°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
.032	.095	.456	.145	.318	Upper	.010	.752	.712	.655	.660	.658
.053	-.101	.206	-.069	.104		.080	.068	.031	-.043	-.119	-.082
.100	-.164	.050	-.195	-.092		.130	-.869	-1.005	-1.145	-1.232	-1.335
.145	-.152	-.012	-.170	-.086		.145	-5.258	-4.839	-4.587	-5.600	-5.485
.189	-.051	.025	-.082	-.031		.155	-1.991	-1.923	-1.960	-2.181	-1.942
.234	-.089	.094	-.031	-.024		.180	-1.405	-1.149	-1.317	-1.420	-1.638
.260	-.082	.112	.013	-.074		.220	-.733	-.712	-.851	-1.131	-1.126
.326	-.063	.125	-.006	.006		.270	-.419	-.481	-.649	-.836	-.892
.371	-.120	.175	-.101	.043		.400	-.080	-.050	-.423	-.786	-.886
.392	.013	.210	-.126	.214		.620	.598	.500	-.447	-1.383	-1.398
.413	-.171	.250	-.339	.061		.685	-.881	-.487	-1.268	-7.007	-9.502
.434	-.228	.275	-.547	.073		.693	-.869	-.687	-2.431	-7.692	-8.742
.457	-.228	.285	-.459	.214		.700	-.709	-.731	-1.776	-4.864	-6.592
.480	-.253	.300	-.302	.325		.720	-.579	-.587	-.692	-2.086	-2.758
.502	-.316	.320	-.157	.349		.750	-.604	-.587	-.637	-1.320	-1.885
.551	-.342	.340	-.075	.441		.800	-.536	-.593	-.557	-.823	-1.430
.585	-.392	.350	-.220	.472		.900	-.524	-.562	-.631	-.459	-1.018
.592	-.481	.337	-.553	-.484		.980	-.518	-.524	-.582	-.075	-.329
.613	-.544	.237	-.603	-.500	Lower	.025	-.018	.194	.269	.295	.164
.634	-.557	.112	-.578	-.502		.120	.049	.194	.269	.239	.082
.655	-.576	-.037	-.534	-.588		.220	.394	.175	.239	.207	.025
.675	-.500	-.194	-.540	-.619		.300	.555	.275	.294	.308	.272
.696	-.500	-.244	-.522	-.619		.620	.518	.687	.631	.647	.304
.774	-.177	-.050	-.365	-.300		.750	.906	.787	.735	.704	.639
.852	.000	.000	-.094	-.165		.850	.616	.637	.594	.635	.607
.930	.025	-.156	-.031	-.104		.950	.296	.256	.245	.471	.418

$\alpha = 13.5°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
.032	-.065	.665	-.076	.251	Upper	.010	.142	-.039	-.295	-.662	-.260
.053	-.227	.439	-.242	.025		.080	-.639	-.652	-.754	-.917	-.832
.100	-.195	.226	-.357	-.182		.130	-1.852	-2.007	-2.143	-2.267	-2.495
.145	-.123	.123	-.293	-.189		.145	-7.208	-6.860	-6.354	-7.520	-7.562
.189	-.084	.161	-.236	-.126		.155	-2.839	-2.936	-2.947	-3.273	-3.002
.234	-.084	.213	-.096	-.170		.180	-1.826	-1.762	-1.736	-2.101	-2.352
.280	-.091	.226	.045	-.176		.220	-.981	-1.110	-1.269	-1.554	-1.592
.326	-.097	.226	.013	-.182		.270	-.613	-.742	-.943	-1.146	-1.267
.371	-.208	.310	-.140	-.163		.400	-.271	-.155	-.553	-.981	-1.156
.392	-.240	.360	-.420	-.094		.620	.245	.207	-.698	-1.758	-1.806
.413	-.279	.413	-.694	.189		.685	-1.123	-.974	-1.100	-7.692	-11.181
.434	-.325	.432	-.891	.478		.693	-.910	-.807	-2.187	-8.062	-10.363
.457	-.318	.432	-.650	.566		.700	-.723	-.729	-1.615	-5.145	-7.848
.480	-.325	.432	-.427	.515		.720	-.600	-.555	-.666	-2.248	-3.417
.502	-.364	.432	-.280	.478		.750	-.594	-.561	-.603	-1.414	-2.352
.551	-.351	.432	-.242	.490		.800	-.568	-.561	-.540	-.917	-1.780
.585	-.416	.432	-.140	.553		.900	-.561	-.568	-.584	-.497	-1.299
.592	-.487	.419	-.579	-.515		.980	-.568	-.561	-.559	-.267	-.598
.613	-.416	.303	-.650	-.277	Lower	.025	.413	.626	.616	.650	.520
.634	-.481	.168	-.599	-.528		.120	.768	.820	.742	.720	.572
.655	-.520	.013	-.548	-.597		.220	.774	.742	.742	.713	.624
.675	-.468	-.161	-.548	-.591		.300	.632	.658	.666	.630	.507
.696	-.513	-.181	-.529	-.584		.620	.594	.820	.792	.713	.149
.774	-.286	.026	-.369	-.302		.750	.897	.962	.804	.739	.611
.852	-.052	.032	-.134	-.226		.850	.652	.703	.610	.637	.513
.930	-.013	-.019	-.083	.013		.950	.310	.316	.245	.439	.299

TABLE 15 Continued
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 6.0 h_d/c = 3.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					α = 19.1°						
.032		.721	-.323	.131	Upper	.010	-2.584	-1.098	-2.204	-2.656	
.053		.507	-.451	-.031		.080	-1.334	-1.156	-1.367	-1.984	
.100		.305	-.551	-.268		.130	-2.544	-2.495	-2.572	-2.871	
.145		.221	-.491	-.275		.145	-8.111	-7.413	-6.868	-8.472	
.189		.227	-.424	-.225		.155	-3.329	-3.437	-3.397	-4.014	
.234		.253	-.202	-.275		.180	-2.126	-2.124	-2.248	-2.649	
.280		.273	.027	-.275		.220	-1.177	-1.358	-1.480	-1.930	
.326		.305	.007	-.331		.270	-.706	-.903	-1.043	-1.425	
.371		.390	-.249	-.343		.400	-.347	-.214	-.537	-1.089	
.392		.450	-.706	-.387		.620	.183	.045	-.731	-1.970	
.413		.513	-.962	.187		.685	-.870	-.494	-.156	-5.030	
.434		.513	-1.419	.556		.693	-.857	-.630	-1.336	-5.379	
.457		.500	-1.029	.624		.700	-.680	-.689	-1.068	-3.140	
.480		.490	-.646	.593		.720	-.530	-.559	-.568	-1.210	
.502		.480	-.417	.556		.750	-.530	-.559	-.568	-.767	
.551		.470	-.276	.531		.800	-.530	-.552	-.556	-.666	
.585		.468	-.155	.599		.900	-.530	-.546	-.587	-.646	
.592		.435	-.585	-.500		.980	-.530	-.533	-.549	-.504	
.613		.286	-.612	-.300							
.634		.169	-.605	-.506	Lower	.025	.661	.773	.774	.814	
.655		-.013	-.605	-.556		.120	.883	.832	.780	.773	
.675		-.156	-.619	-.556		.220	.850	.799	.768	.773	
.696		-.156	-.572	-.581		.300	.713	.708	.681	.706	
.774		.032	-.336	-.430		.620	.713	.851	.787	.719	
.852		.019	-.182	-.275		.750	.903	.903	.768	.740	
.930		.110	-.182	.131		.850	.700	.689	.618	.639	
						.950	.379	.318	.275	.356	
					α = 23.2°						
.032	-.250	.731	-.481	.066	Upper	.010	-6.077	-4.128	-3.263	-3.218	-3.174
.053	-.362	.528	-.588	-.113		.060	-1.609	-2.627	-3.236	-3.172	-1.758
.100	-.231	.386	-.668	-.358		.130	-2.635	-2.573	-2.586	-2.738	-2.608
.145	-.191	.278	-.588	-.405		.145	-7.737	-7.313	-6.499	-6.464	-6.915
.189	-.092	.291	-.541	-.345		.155	-3.141	-3.636	-3.448	-3.339	-2.891
.234	-.066	.332	-.207	-.411		.180	-1.981	-2.323	-2.361	-2.237	-2.259
.280	-.105	.379	.053	-.424		.220	-1.212	-1.503	-1.519	-1.549	-1.436
.326	-.184	.393	-.020	-.524		.270	-1.032	-.995	-1.054	-1.115	-1.093
.371	-.382	.467	-.374	-.550		.400	-.519	-.311	-.511	-.681	-1.199
.392	-.450	.515	-.821	-.663		.620	-.071	.014	-.623	-.948	-1.067
.413	-.533	.569	-1.042	.239		.685	-.936	-.494	.186	-1.696	-1.930
.434	-.566	.616	-1.729	.637		.693	-.949	-.731	-1.127	-1.963	-1.699
.457	-.520	.590	-1.175	.716		.700	-.756	-.779	-.995	-1.149	-1.475
.480	-.468	.565	-.881	.676		.720	-.628	-.650	-.610	-.601	-.988
.502	-.461	.530	-.574	.630		.750	-.628	-.650	-.623	-.634	-.988
.551	-.362	.510	-.294	.584		.800	-.667	-.697	-.623	-.674	-.988
.585	-.369	.501	-.287	.643		.900	-.622	-.596	-.663	-.688	-.962
.592	-.454	.433	-.621	-.590		.980	-.628	-.623	-.617	-.634	-.843
.613	-.369	.284	-.654	-.436							
.634	-.448	.163	-.694	-.610	Lower	.025	.737	.894	.862	.861	.718
.655	-.448	-.014	-.694	-.683		.120	.878	.894	.816	.815	.632
.675	-.369	-.183	-.668	-.696		.220	.840	.846	.836	.815	.672
.696	-.316	-.183	-.614	-.703		.300	.744	.772	.763	.768	.632
.774	-.198	.095	-.214	-.495		.620	.763	.914	.836	.768	.290
.852	-.105	.041	-.080	-.285		.750	.865	.955	.836	.775	.626
.930	-.145	.169	-.234	.206		.850	.667	.745	.650	.654	.527
						.950	.391	.352	.292	.300	.224

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TABLE 15 Continued
(e)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 8.0$ $h_d/c = 4.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing , flap , or aileron				
					α = -1.2°						
.032	.276	.293	.296	.299	Upper	.010	.962	.879	.809	.816	.778
.053	.055	.070	.067	.070		.080	.625	.575	.463	.417	.404
.100	-.092	-.023	-.073	-.064		.130	.006	-.082	-.240	-.296	-.447
.145	-.080	-.082	-.054	-.035		.145	-3.056	-2.591	-2.527	-3.387	-3.399
.189	-.012	-.012	.000	-.006		.155	-.897	-.762	-.862	-1.034	-.925
.234	-.018	.059	-.006	.012		.180	-.613	-.352	-.569	-.659	-.913
.280	-.031	.070	-.054	.041		.220	-.212	-.129	-.328	-.611	-.588
.326	.006	.082	-.036	.082		.270	-.047	-.006	-.217	-.375	-.447
.371	-.018	.117	.006	.158		.400	.224	.293	-.106	-.460	-.465
.392	-.073	.100	.025	.340		.620	1.062	.903	-.346	-.732	-.815
.413	-.031	.117	.054	-.264		.685	-.755	-.352	-.287	-5.509	-5.965
.434	-.067	.123	-.067	-.410		.693	-.743	-.586	-1.348	-5.993	-5.102
.457	-.067	.125	-.073	-.340		.700	-.596	-.616	-1.073	-3.574	-3.485
.480	-.129	.130	.018	-.211		.720	-.484	-.463	-.569	-1.343	-1.041
.502	-.190	.135	.127	-.059		.750	-.460	-.469	-.539	-.804	-.643
.551	-.294	.140	.000	.217		.800	-.460	-.504	-.516	-.460	-.582
.585	-.386	.141	-.393	.229		.900	-.448	-.457	-.557	-.308	-.563
.592	-.508	.164	-.502	-.422		.980	-.448	-.422	-.475	-.115	-.423
.613	-.570	.070	-.490	-.188							
.634	-.521	.018	-.490	-.410	Lower	.025	-.366	-.317	-.094	-.042	-.043
.655	-.508	-.059	-.502	-.481		.120	-.437	-.270	-.100	-.054	-.067
.675	-.484	-.188	-.502	-.516		.220	-.413	-.287	-.100	-.067	-.067
.696	-.521	-.211	-.490	-.557		.300	-.301	-.328	-.123	-.079	-.092
.774	-.153	-.064	-.321	-.299		.620	.519	.334	-.129	-.187	-.355
.852	.067	-.053	-.018	-.100		.750	.861	.510	-.035	-.091	-.129
.930	.055	-.147	.030	-.141		.850	.684	.627	.217	-.006	.031
						.950	.330	.358	.229	.181	.190
					α = 6.1°						
.032	.119	.456	.143	.321	Upper	.010	.780	.752	.701	.720	.693
.053	-.066	.216	-.062	.103		.080	.112	.111	.048	-.037	-.006
.100	-.161	.055	-.211	-.079		.130	-.799	-.838	-.992	-1.117	-1.135
.145	-.108	-.012	-.155	-.054		.145	-5.039	-4.321	-4.155	-5.279	-4.903
.189	-.036	.037	-.093	-.012		.155	-1.873	-1.633	-1.699	-2.022	-1.666
.234	-.054	.117	-.019	-.006		.180	-1.299	-.937	-1.137	-1.309	-1.427
.280	-.030	.129	.006	-.006		.220	-.612	-.512	-.701	-1.055	-.973
.326	-.036	.136	.000	.042		.270	-.331	-.277	-.490	-.738	-.747
.371	-.090	.191	-.112	.085		.400	.087	.191	-.266	-.726	-.747
.392	-.110	.210	-.112	.073		.620	1.011	.838	-.490	-1.303	-1.224
.413	-.137	.234	-.310	.060		.685	-.887	-.450	-.429	-6.433	-8.773
.434	-.161	.271	-.484	.067		.693	-.899	-.826	-1.615	-7.196	-8.057
.457	-.161	.280	-.409	.194		.700	-.718	-.746	-1.294	-4.355	-6.062
.480	-.209	.290	-.186	.302		.720	-.599	-.579	-.659	-1.836	-2.461
.502	-.245	.300	.000	.357		.750	-.593	-.586	-.605	-1.123	-1.642
.551	-.323	.315	-.149	.441		.800	-.593	-.604	-.575	-.689	-1.224
.585	-.406	.333	-.533	.478		.900	-.587	-.573	-.617	-.465	-.830
.592	-.561	.327	-.602	-.532		.980	-.587	-.567	-.562	-.199	-.227
.613	-.657	.203	-.608	-.525							
.634	-.579	.099	-.602	-.550	Lower	.025	-.062	.160	.254	.261	.173
.655	-.508	-.055	-.571	-.611		.120	-.012	.154	.248	.248	.078
.675	-.448	-.210	-.596	-.593		.220	.293	.136	.194	.186	.036
.696	-.526	-.259	-.596	-.611		.300	.506	.222	.224	.186	.161
.774	-.293	-.062	-.459	-.333		.620	.400	.666	.623	.614	.317
.852	.000	.018	-.136	-.163		.750	.912	.789	.695	.664	.633
.930	.054	-.136	-.037	-.067		.850	.618	.653	.581	.633	.609
						.950	.268	.277	.230	.434	.436
					α = 13.5°						
.032	-.057	.607	-.095	.267	Upper	.010	.200	.110		-.171	-.101
.053	-.234	.381	-.259	.051		.080	-.549	-.542	-.624	-.797	-.746
.100	-.183	.200	-.342	-.166		.130	-1.723	-1.800	-1.942	-2.119	-2.347
.145	-.133	.123	-.304	-.178		.145	-6.679	-6.363	-5.948	-7.243	-7.212
.189	-.089	.142	-.234	-.115		.155	-2.549	-2.646	-2.655	-3.074	-2.809
.234	-.089	.194	-.082	-.153		.180	-1.665	-1.568	-1.732	-1.955	-2.208
.280	-.089	.213	.032	-.146		.220	-.865	-.955	-1.095	-1.468	-1.518
.326	-.089	.232	.013	-.166		.270	-.523	-.594	-.783	-1.063	-1.177
.371	-.190	.303	-.145	-.134		.400	-.187	.065	-.388	-.879	-1.082
.392	-.220	.350	-.386	-.045		.620	.800	.671	-.713	-1.676	-1.689
.413	-.253	.394	-.664	.217		.685	-.897	-.523	-.382	-6.649	-9.970
.434	-.291	.439	-.841	.478		.693	-.884	-.871	-1.624	-7.382	-9.957
.457	-.272	.440	-.626	.560		.700	-.703	-.800	-1.274	-4.517	-7.585
.480	-.278	.435	-.373	.529		.720	-.574	-.626	-.688	-1.872	-3.233
.502	-.329	.430	-.171	.471		.750	-.568	-.639	-.637	-1.139	-2.214
.551	-.342	.420	-.190	.484		.800	-.574	-.639	-.611	-.702	-1.626
.585	-.392	.413	-.538	.560		.900	-.620	-.632	-.643	-.563	-1.126
.592	-.462	.387	-.588	-.573		.980	-.607	-.626	-.618	-.354	-.443
.613	-.462	.284	-.626	-.344							
.634	-.569	.136	-.601	-.599	Lower	.025	.381	.600	.586	.614	.506
.655	-.550	-.026	-.595	-.662		.120	.781	.787	.726	.715	.582
.675	-.506	-.194	-.595	-.643		.220	.768	.755	.751	.734	.626
.696	-.544	-.213	-.563	-.637		.300	.665	.665	.669	.639	.506
.774	-.342	.019	-.462	-.401		.620	.600	.723	.771	.702	.177
.852	-.051	.032	-.190	-.274		.750	.858	.968	.815	.778	.614
.930	.013	-.058	-.139	.013		.850	.658	.716	.637	.626	.512
						.950	.329	.265	.242	.367	.348

TABLE 15 Concluded
(e) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 8.0 h_d/c = 4.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 19.2°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.181	.608	-.323	.105
.053	-.297	.392	-.439	-.105
.100	-.200	.275	-.484	-.316
.145	-.174	.170	-.445	-.323
.189	-.077	.183	-.400	-.277
.234	-.077	.222	-.142	-.303
.280	-.110	.268	.019	-.310
.326	-.090	.301	.013	-.356
.371	-.271	.386	-.207	-.382
.392	-.320	.440	-.613	-.415
.413	-.374	.497	-.839	.158
.434	-.426	.549	-1.291	.540
.457	-.387	.550	-.916	.632
.480	-.374	.530	-.542	.599
.502	-.394	.510	-.297	.540
.551	-.361	.490	-.323	.547
.585	-.381	.471	-.536	.580
.592	-.439	.379	-.568	-.514
.613	-.387	.249	-.594	-.525
.634	-.529	.144	-.568	-.553
.655	-.549	-.007	-.561	-.619
.675	-.497	-.170	-.574	-.626
.696	-.478	-.190	-.568	-.639
.774	-.271	.039	-.400	-.500
.852	-.142	.013	-.194	-.389
.930	-.077	.150	-.213	.132

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-1.847	-1.923	-2.147	-2.336	-2.084
	.080	-1.126	-.962	-1.251	-1.568	-1.252
	.130	-2.265	-2.374	-2.621	-2.697	-2.969
	.145	-7.313	-7.077	-6.955	-7.963	-8.131
	.155	-2.948	-3.212	-3.411	-3.704	-3.446
	.180	-1.847	-1.923	-2.239	-2.401	-2.659
	.220	-.981	-1.190	-1.455	-1.755	-1.800
	.270	-.569	-.726	-1.008	-1.265	-1.420
	.400	-.259	-039	-.421	-.955	-1.278
	.620	.512	301	-.863	-1.742	-1.755
	.685	-.791	- 399	-.046	-4.672	-10.351
	.693	-.797	- 693	-1.383	-4.892	-9.499
	.700	-.652	- 726	-1.146	-2.930	-7.189
	.720	-.531	- 536	-.645	-1.104	-2.981
	.750	-.493	- 536	-.632	-.774	-1.949
	.800	-.500	- 549	-.632	-.697	-1.362
	.900	-.544	- 549	-.672	-.620	-.878
	.980	-.544	- 530	-.619	-.516	-.336
Lower	.025	.626	772	.764	.774	.639
	.120	.854	837	.764	.742	.561
	.220	.803	.818	.777	.761	.620
	.300	.702	.726	.698	.678	.542
	.620	.633	.798	.771	.697	.155
	.750	.898	.968	.784	.723	.594
	.850	.683	720	.599	.613	.523
	.950	.361	340	.263	.336	.374

α = 23.2°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.206	.677	-.458	.039
.053	-.287	.443	-.549	-.144
.100	-.162	.342	-.600	-.366
.145	-.137	.240	-.549	-.399
.189	-.050	.253	-.490	-.327
.234	-.025	.297	-.174	-.392
.280	-.062	.329	.045	-.392
.326	-.137	.361	.006	-.484
.371	-.325	.424	-.342	-.523
.392	-.395	.475	-.736	-.628
.413	-.481	.531	-.916	.196
.434	-.512	.588	-1.523	.628
.457	-.462	.580	-1.052	.687
.480	-.400	.550	-.807	.641
.502	-.393	.510	-.465	.589
.551	-.317	.480	-.342	.589
.585	-.343	.468	-.458	.641
.592	-.412	.418	-.574	-.523
.613	-.325	.291	-.600	-.500
.634	-.437	.152	-.626	-.556
.655	-.437	-.019	-.632	-.648
.675	-.375	-.164	-.620	-.628
.696	-.312	-.177	-.574	-.661
.774	-.187	.089	-.277	-.560
.852	-.106	.032	-.123	-.327
.930	-.094	.177	-.213	.190

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-5.195	-2 771	-3.172	-3.291	-3.047
	.080	-1.613	-2 265	-3.081	-3.265	-2.335
	.130	-2.439	-2 252	-2.505	-2.704	-2.760
	.145	-7.163	-6 579	-6.384	-7.131	-7.524
	.155	-2.839	-3 188	-3.310	-3.594	-3.315
	.180	-1.768	-1 993	-2.276	-2.459	-2.616
	.220	-1.162	-1 253	-1.452	-1.749	-1.792
	.270	-1.007	- 791	-.988	-1.265	-1.449
	.400	-.413	- 101	-.451	-.858	-1.361
	.620	.207	127	-.589	-1.375	-1.280
	.685	-.865	- 399	.249	-2.201	-5.669
	.693	-.910	- 683	-1.138	-2.317	-4.976
	.700	-.736	- 709	-.994	-1.510	-3.615
	.720	-.613	- 550	-.608	-.800	-1.299
	.750	-.671	- 569	-.608	-.768	-.843
	.800	-.626	- 550	-.608	-.736	-.662
	.900	-.600	- 531	-.680	-.678	-.656
	.980	-.587	- 531	-.628	-.620	-.468
Lower	.025	.736	854	.850	.832	.687
	.120	.891	867	.785	.755	.599
	.220	.845	822	.824	.794	.662
	.300	.761	753	.752	.742	.612
	.620	.639	822	.798	.723	.281
	.750	.949	943	.824	.729	.662
	.850	.697	740	.628	.626	.574
	.950	.400	361	.275	.316	.362

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TABLE 16 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 2.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = -1.4°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.262	.293	.298	.314
.053	.067	.062	.068	.074
.100	-.079	-.037	-.093	-.074
.145	-.073	-.094	-.068	-.037
.189	.006	-.037	.000	-.012
.234	-.030	.037	-.031	.012
.280	-.043	.050	-.050	.031
.326	-.018	.075	-.031	.074
.371	-.067	.112	.074	.148
.392	-.092	.116	.037	.345
.413	-.116	.119	-.105	-.148
.434	-.158	.131	-.335	-.284
.457	-.176	.149	-.409	-.191
.480	-.243	.168	-.372	-.049
.502	-.310	.186	-.385	.111
.551	-.347	.222	-.484	.290
.585	-.347	.250	-.558	.247
.592	-.335	.225	-.596	-.468
.613	-.286	.206	-.453	-.536
.634	-.250	.137	-.335	-.604
.655	-.201	.069	-.273	-.203
.675	-.128	-.025	-.199	-.129
.696	-.091	-.050	-.149	-.080
.774	.012	-.043	-.037	-.099
.852	-.018	-.037	.000	-.117
.930	.055	-.150	.074	-.160

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.955	.862	.826	.837	.797
	.080	.471	.362	.314	.323	.353
	.130	-.298	-.537	-.604	-.571	-.560
	.145	-4.094	-4.008	-3.649	-4.200	-3.743
	.155	-1.476	-1.548	-1.479	-1.464	-1.150
	.180	-1.092	-.962	-1.036	-.949	-1.059
	.220	-.633	-.687	-.703	-.794	-.651
	.270	-.484	-.587	-.598	-.552	-.493
	.400	-.471	-.487	-.493	-.552	-.505
	.620	-.782	-.368	-.271	-.775	-.834
	.685	-4.646	-1.055	-.086	-2.407	-6.116
	.693	-4.504	-.949	-1.344	-2.277	-5.179
	.700	-2.667	-.943	-1.183	-1.489	-3.627
	.720	-1.104	-.837	-.814	-.806	-1.156
	.750	-.664	-.549	-.801	-.825	-.755
	.800	-.490	-.475	-.678	-.782	-.688
	.900	-.521	-.500	-.604	-.682	-.615
	.980	-.422	-.456	-.592	-.602	-.438
Lower	.025	-.261	-.169	.012	.087	-.030
	.120	-.329	-.144	.006	.025	-.061
	.220	-.285	-.150	-.037	.006	-.049
	.300	-.099	-.206	-.099	-.050	-.091
	.620	.490	.275	.006	-.037	-.183
	.750	.775	.531	.074	.087	.116
	.850	.701	.599	.277	.205	.292
	.950	.409	.350	.216	.186	.298

α = 5.8°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.081	.459	.138	.329
.053	-.112	.239	-.069	.089
.100	-.194	.069	-.195	-.101
.145	-.144	-.006	-.163	-.070
.189	-.081	.044	-.107	-.025
.234	-.100	.107	-.057	-.013
.280	-.100	.126	.031	-.025
.326	-.100	.138	.000	-.013
.371	-.175	.207	-.113	.038
.392	-.219	.242	-.220	.215
.413	-.262	.277	-.484	.177
.434	-.318	.308	-.754	.177
.457	-.337	.322	-.704	.310
.480	-.418	.337	-.603	.392
.502	-.462	.352	-.559	.443
.551	-.450	.381	-.584	.462
.585	-.443	.402	-.654	.512
.592	-.425	.390	-.698	-.702
.613	-.350	.308	-.540	-.506
.634	-.312	.201	-.390	-.481
.655	-.281	.101	-.327	-.215
.675	-.194	-.025	-.264	-.152
.696	-.150	-.038	-.176	-.089
.774	.012	.000	-.079	.057
.852	-.031	.038	.019	-.082
.930	.044	-.119	.075	-.120

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.667	.647	.582	.591	.593
	.080	-.115	-.126	-.196	-.239	-.144
	.130	-1.218	-1.345	-1.512	-1.458	-1.455
	.145	-6.372	-5.907	-5.548	-6.134	-5.663
	.155	-2.558	-2.520	-2.499	-2.476	-2.117
	.180	-1.846	-1.603	-1.708	-1.609	-1.711
	.220	-1.128	-1.112	-1.177	-1.257	-1.136
	.270	-.821	-.899	-.949	-.930	-.899
	.400	-.667	-.647	-.702	-.754	-.812
	.620	-.910	-.503	-.405	-.855	-1.149
	.685	-4.628	-1.420	-.215	-3.281	-8.242
	.693	-4.218	-.848	-1.379	-2.476	-7.455
	.700	-2.417	-.911	-1.208	-1.760	-5.507
	.720	-1.000	-.811	-.810	-.974	-2.223
	.750	-.686	-.704	-.930	-.874	-1.505
	.800	-.583	-.522	-.607	-.836	-1.168
	.900	-.647	-.566	-.607	-.698	-.968
	.980	-.513	-.540	-.582	-.603	-.475
Lower	.025	.071	.333	.335	.352	.194
	.120	.141	.283	.310	.277	.100
	.220	.526	.283	.266	.239	.100
	.300	.609	.459	.449	.452	.450
	.620	.705	.679	.671	.641	.287
	.750	.827	.735	.677	.679	.599
	.850	.603	.572	.563	.566	.512
	.950	.359	.207	.202	.239	.300

α = 13.3°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.099	.632	-.103	.262
.053	-.245	.419	-.250	.039
.100	-.206	.207	-.353	-.196
.145	-.153	.116	-.321	-.203
.189	-.099	.142	-.256	-.157
.234	-.119	.207	-.096	-.183
.280	-.133	.207	.077	-.196
.326	-.139	.219	.038	-.235
.371	-.272	.290	-.199	-.216
.392	-.325	.355	-.538	-.170
.413	-.378	.419	-.910	.196
.434	-.418	.452	-1.205	.517
.457	-.431	.452	-.949	.602
.480	-.444	.452	-.814	.563
.502	-.524	.452	-.718	.510
.551	-.464	.452	-.744	.549
.585	-.438	.452	-.756	.602
.592	-.411	.439	-.769	-.785
.613	-.332	.336	-.590	-.687
.634	-.298	.232	-.449	-.419
.655	-.272	.103	-.346	-.196
.675	-.179	.000	-.256	-.124
.696	-.126	.000	-.154	-.052
.774	-.079	.039	-.077	-.042
.852	-.033	.077	.000	-.033
.930	.027	.000	.045	-.026

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.090	-1.020	-1.269	-1.154	-.259
	.080	-.872	-.897	-1.033	-1.038	-.869
	.130	-2.231	-2.465	-2.682	-2.481	-2.580
	.145	-8.096	-8.015	-7.548	-8.064	-7.825
	.155	-3.359	-3.620	-3.663	-3.558	-3.203
	.180	-2.269	-2.310	-2.459	-2.295	-2.420
	.220	-1.410	-1.600	-1.694	-1.705	-1.618
	.270	-1.026	-1.252	-1.315	-1.231	-1.253
	.400	-.821	-.858	-.896	-.917	-1.081
	.620	-.987	-.697	-.641	-.968	-1.499
	.685	-3.962	-1.878	-.844	-4.212	-9.456
	.693	-3.212	-.652	-1.308	-1.635	-8.554
	.700	-2.000	-.942	-1.119	-1.218	-6.320
	.720	-1.026	-.794	-.772	-.808	-2.546
	.750	-.782	-.813	-.726	-.756	-1.678
	.800	-.660	-.684	-.746	-.712	-1.293
	.900	-.641	-.658	-.582	-.615	-1.121
	.980	-.494	-.658	-.602	-.615	-.690
Lower	.025	.462	.684	.667	.679	.550
	.120	.776	.787	.746	.718	.584
	.220	.750	.736	.726	.699	.637
	.300	.654	.691	.654	.635	.524
	.620	.718	.774	.739	.686	.206
	.750	.827	.965	.739	.718	.603
	.850	.577	.632	.536	.577	.491
	.950	.385	.252	.196	.250	.239

TABLE 16 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 2.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 19.0°											
.032	-.190	.751	-.298	.150	Upper	.010	-4.014	- .766	-2.924	-2.918	-2.499
.053	-.314	.547	-.431	-.026		.080	-1.715	- .772	-2.499	-2.573	-1.105
.100	-.203	.323	-.517	-.262		.130	-3.127	-2.918	-2.924	-2.845	-2.649
.145	-.177	.231	-.458	-.301		.145	-9.736	-8.693	-7.758	-8.103	-7.182
.189	-.092	.263	-.411	-.235		.155	-4.162	-4.215	-4.049	-3.926	-2.950
.234	-.098	.316	-.159	-.288		.180	-2.764	-2.779	-2.813	-2.626	-2.185
.280	-.124	.310	.099	-.314		.220	-1.701	-1.936	-1.956	-1.870	-1.354
.326	-.164	.323	.053	-.392		.270	-1.231	-1.515	-1.504	-1.393	-.988
.371	-.347	.415	-.345	-.445		.400	-.941	-1.034	-1.007	-.981	-1.053
.392	-.412	.478	-.809	-.530		.620	-1.002	-.711	-.700	-.935	-.909
.413	-.477	.540	-1.094	.183		.685	-3.718	-1.133	-.458	-2.679	-1.760
.434	-.543	.573	-1.698	.615		.693	-3.839	- .218	-1.779	-3.183	-1.426
.457	-.504	.565	-1.353	.693		.700	-2.508	- .218	-1.537	-1.764	-1.197
.480	-.497	.556	-1.074	.654		.720	-1.352	- .014	-1.053	-.915	-.844
.502	-.530	.548	-.962	.608		.750	-1.002	.869	-.870	-.968	-.890
.551	-.438	.532	-.948	.576		.800	-.753	.751	-.720	-.902	-.883
.585	-.399	.520	-1.008	.621		.900	-.531	.751	-.680	-.875	-.883
.592	-.373	.487	-1.214	-1.190		.980	-.410	.777	-.667	-.802	-.870
.613	-.275	.382	-1.068	-.800							
.634	-.235	.263	-.690	-.654	Lower	.025	.719	.830	.811	.849	.687
.655	-.216	.125	-.405	-.288		.120	.894	.843	.765	.756	.615
.675	-.137	.007	-.232	-.105		.220	.841	.810	.805	.769	.648
.696	-.085	.033	-.126	-.033		.300	.760	.764	.720	.729	.589
.774	-.081	.060	-.073	-.017		.620	.787	.803	.746	.723	.307
.852	-.078	.086	-.020	.000		.750	.861	.856	.785	.749	.602
.930	-.013	.079	.000	.065		.850	.686	.672	.569	.610	.484
						.950	.471	.296	.229	.245	.150
α = 23.0°											
.032	-.252	.789	-.484	.020	Upper	.010	-8.464	- .820	-4.081	-3.820	-3.548
.053	-.351	.584	-.570	-.141		.080	-1.969	- .554	-4.176	-3.846	-2.347
.100	-.219	.398	-.637	-.363		.130	-3.225	- .885	-3.281	-3.084	-2.354
.145	-.153	.279	-.603	-.410		.145	-9.360	- .971	-6.536	-6.492	-6.280
.189	-.080	.318	-.537	-.356		.155	-3.925	- .231	-3.866	-3.541	-2.725
.234	-.066	.351	-.179	-.417		.180	-2.590	- .898	-2.851	-2.546	-2.109
.280	-.119	.358	.126	-.437		.220	-1.629	- .069	-2.024	-1.850	-1.326
.326	-.199	.371	.027	-.538		.270	-1.380	- .618	-1.580	-1.439	-1.001
.371	-.405	.477	-.438	-.619		.400	-.988	- .127	-1.163	-1.029	-.962
.392	-.498	.537	-.988	-.820		.620	-.929	.889	-1.015	-1.001	-.935
.413	-.590	.597	-1.267	.175		.685	-3.879	- .393	-.363	-1.757	-1.817
.434	-.623	.610	-2.042	.652		.693	-4.127	- .519	-1.634	-1.930	-1.439
.457	-.550	.598	-1.532	.726		.700	-2.701	- .233	-1.452	-1.320	-1.253
.480	-.484	.586	-1.293	.686		.720	-1.426	.869	-.908	-.922	-.889
.502	-.471	.574	-1.121	.646		.750	-.968	.729	-.874	-.915	-.915
.551	-.298	.550	-1.061	.612		.800	-.706	.670	-.861	-.869	-.889
.585	-.239	.537	-1.147	.646		.900	-.445	.656	-.807	-.855	-.842
.592	-.219	.491	-1.519	-1.466		.980	-.406	.683	-.753	-.842	-.776
.613	-.166	.371	-1.326	-.950							
.634	-.153	.272	-.789	-.646	Lower	.025	.791	.889	.867	.822	.696
.655	-.126	.133	-.458	-.323		.120	.890	.849	.780	.763	.623
.675	-.066	-.007	-.279	-.121		.220	.857	.816	.827	.756	.656
.696	-.046	.046	-.153	-.027		.300	.798	.796	.753	.749	.610
.774	-.027	.076	.007	-.007		.620	.791	.849	.773	.696	.305
.852	-.106	.106	-.073	.013		.750	.863	.889	.814	.723	.603
.930	-.033	.119	-.033	.128		.850	.680	.710	.598	.584	.511
						.950	.504	.371	.269	.252	.192

L-1721

L-927

TABLE 16 Continued
(b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 1.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = -1.7°

Fuselage:

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.254	.335	.292	.285
.053	.043	.070	.074	.063
.100	-.105	-.038	-.081	-.089
.145	-.087	-.089	-.056	-.051
.189	-.019	-.019	.000	.000
.234	-.056	.051	.019	.006
.280	-.062	.082	-.019	.025
.326	-.050	.089	-.006	.063
.371	-.093	.127	-.056	.139
.392	-.130	.149	.006	.342
.413	-.167	.171	-.174	.044
.434	-.230	.190	-.403	-.101
.457	-.273	.211	-.533	-.070
.480	-.329	.233	-.496	.051
.502	-.422	.254	-.540	.133
.551	-.447	.296	-.720	.297
.585	-.440	.329	-.875	.247
.592	-.416	.310	-.999	-.886
.613	-.323	.234	-.806	-.851
.634	-.261	.183	-.583	-.816
.655	-.211	.114	-.385	-.171
.675	-.130	.051	-.242	-.089
.696	-.068	.025	-.155	-.025
.774	-.068	.070	-.056	.000
.852	-.031	-.025	.006	-.152
.930	.056	-.202	.093	-.278

Wing, flap, or aileron:

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.924	.835	.797	.819	.794
	.080	.377	.278	.215	.192	.267
	.130	-.490	-.753	-.879	-.794	-.763
	.145	-4.682	-4.567	-4.415	-4.857	-4.311
	.155	-1.772	-1.885	-1.923	-1.836	-1.458
	.180	-1.345	-1.246	-1.404	-1.247	-1.309
	.220	-.817	-.949	-1.050	-1.061	-.868
	.270	-.672	-.860	-.949	-.825	-.713
	.400	-.729	-.879	-1.025	-.912	-.763
	.620	-1.269	-1.480	-1.518	-1.867	-1.185
	.685	-6.624	-4.694	-3.966	-7.314	-5.986
	.693	-6.567	-5.851	-6.010	-8.195	-4.932
	.700	-4.179	-4.378	-4.795	-5.000	-3.716
	.720	-2.036	-2.050	-2.208	-2.320	-1.538
	.750	-1.383	-1.189	-1.316	-1.458	-1.185
	.800	-.918	-.601	-.683	-.825	-1.024
	.900	-.597	-.164	-.101	-.372	-.862
	.980	-.031	-.044	.095	-.006	-.670
Lower	.025	-.126	-.013	.145	.205	.012
	.120	-.176	-.006	.139	.155	-.037
	.220	-.145	-.013	.114	.112	-.025
	.300	.025	-.051	.038	.074	-.068
	.620	.515	.367	.164	.099	-.11[illegible]
	.750	.773	.557	.278	.236	.180
	.850	.748	.671	.405	.354	.285
	.950	.566	.525	.430	.391	.205

α = 5.5°

Fuselage:

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.071	.481	.141	.321
.053	-.110	.253	-.071	.077
.100	-.208	.089	-.224	-.115
.145	-.156	.025	-.167	-.077
.189	-.071	.063	-.122	-.045
.234	-.104	.114	-.026	-.051
.280	-.104	.127	.064	-.038
.326	-.104	.152	.026	-.038
.371	-.221	.221	-.167	.000
.392	-.273	.263	-.282	.160
.413	-.325	.304	-.577	.231
.434	-.390	.354	-.897	.256
.457	-.422	.366	-.833	.372
.480	-.474	.379	-.756	.442
.502	-.572	.390	-.724	.455
.551	-.533	.416	-.795	.500
.585	-.487	.436	-.853	.545
.592	-.461	.424	-.827	-.808
.613	-.351	.329	-.654	-.650
.634	-.305	.234	-.500	-.551
.655	-.260	.114	-.359	-.487
.675	-.143	.019	-.218	-.179
.696	-.104	.013	-.115	-.038
.774	-.026	.076	-.064	.006
.852	-.052	.038	-.013	-.135
.930	.032	-.114	.058	-.141

Wing, flap, or aileron:

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.538	.493	.417	.449	.513
	.080	-.231	-.316	-.423	-.468	-.351
	.130	-1.423	-1.695	-1.923	-1.872	-1.891
	.145	-6.904	-6.731	-6.564	-7.308	-6.906
	.155	-2.846	-2.980	-3.141	-3.128	-2.768
	.180	-2.083	-1.936	-2.218	-2.103	-2.235
	.220	-1.314	-1.430	-1.641	-1.712	-1.572
	.270	-1.013	-1.227	-1.397	-1.346	-1.299
	.400	-.897	-1.088	-1.308	-1.346	-1.312
	.620	-1.141	-1.581	-1.724	-2.327	-2.008
	.685	-3.885	-4.567	-4.115	-8.923	-11.811
	.693	-3.500	-5.636	-6.026	-9.859	-10.876
	.700	-2.006	-4.175	-4.782	-6.295	-8.355
	.720	-.891	-1.910	-2.244	-3.032	-3.911
	.750	-.756	-1.113	-1.359	-1.955	-2.846
	.800	-.667	-.582	-.769	-1.218	-2.254
	.900	-.558	-.253	-.359	-.545	-1.566
	.980	-.455	-.101	-.179	-.051	-.591
Lower	.025	.167	.436	.372	.436	.221
	.120	.237	.392	.385	.346	.104
	.220	.538	.493	.538	.474	.364
	.300	.615	.607	.628	.590	.507
	.620	.699	.721	.692	.660	.097
	.750	.808	.778	.712	.692	.552
	.850	.590	.645	.615	.609	.481
	.950	.372	.443	.397	.462	.299

α = 13.0°

Fuselage:

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.079	.632	-.113	.277
.053	-.250	.402	-.285	.058
.100	-.191	.191	-.398	-.168
.145	-.145	.112	-.351	-.181
.189	-.105	.138	-.285	-.123
.234	-.112	.198	-.119	-.161
.280	-.138	.217	.066	-.187
.326	-.151	.231	.046	-.239
.371	-.296	.323	-.245	-.239
.392	-.352	.382	-.650	-.213
.413	-.408	.441	-1.041	.213
.434	-.474	.481	-1.379	.574
.457	-.474	.485	-1.141	.632
.480	-.507	.489	-1.001	.600
.502	-.566	.493	-.922	.542
.551	-.487	.501	-.942	.587
.585	-.454	.507	-.902	.639
.592	-.421	.487	-.895	-.858
.613	-.323	.382	-.710	-.465
.634	-.290	.270	-.524	-.348
.655	-.250	.132	-.338	-.374
.675	-.158	.007	-.206	-.258
.696	-.092	.040	-.113	-.077
.774	-.013	.050	.073	-.061
.852	-.066	.059	-.033	-.045
.930	.000	.026	-.007	.032

Wing, flap, or aileron:

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.295	-1.482	-1.781	-2.023	-1.666
	.080	-1.045	-1.120	-1.187	-1.379	-1.232
	.130	-2.532	-2.865	-3.104	-3.203	-3.240
	.145	-8.731	-8.996	-8.447	-9.728	-9.141
	.155	-3.718	-4.169	-4.279	-4.589	-3.978
	.180	-2.545	-2.733	-2.988	-3.064	-3.056
	.220	-1.654	-1.969	-2.142	-2.354	-2.160
	.270	-1.256	-1.614	-1.768	-1.863	-1.778
	.400	-1.064	-1.291	-1.471	-1.651	-1.693
	.620	-1.154	-1.732	-1.826	-2.540	-2.542
	.685	-2.705	-4.445	-3.659	-9.138	-14.700
	.693	-2.263	-5.242	-5.337	-9.920	-13.764
	.700	-1.404	-3.780	-4.220	-6.227	-10.676
	.720	-.654	-1.893	-1.942	-3.011	-5.084
	.750	-.590	-.968	-1.174	-1.916	-3.649
	.800	-.474	-.507	-.729	-1.141	-2.766
	.900	-.410	-.257	-.510	-.564	-1.752
	.980	-.372	-.099	-.348	-.418	-.566
Lower	.025	.500	.731	.716	.736	.566
	.120	.795	.830	.761	.723	.520
	.220	.756	.764	.761	.736	.547
	.300	.667	.711	.691	.643	.461
	.620	.737	.810	.761	.676	-.066
	.750	.821	.902	.742	.716	.487
	.850	.622	.705	.645	.623	.421
	.950	.397	.481	.394	.431	.290

TABLE 16 Concluded
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = .0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.476	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 18.7°											
.032	-.173	.736	-.342	.126	Upper	.010	-5.354	-3.146	-3.727	-3.807	-3.564
.053	-.301	.531	-.461	-.053		.080	-1.864	-2.1[illegible]2	-3.481	-3.708	-2.346
.100	-.212	.318	-.533	-.292		.130	-3.293	-3.1[illegible]7	-3.349	-3.444	-3.590
.145	-.186	.225	-.487	-.332		.145	-10.037	-9.2[illegible]7	-8.687	-9.200	-9.583
.189	-.096	.252	-.448	-.265		.155	-4.334	-4.635	-4.788	-4.847	-4.494
.234	-.083	.305	-.138	-.332		.180	-2.904	-3.110	-3.448	-3.425	-3.513
.280	-.135	.305	.105	-.351		.220	-1.798	-2.255	-2.533	-2.601	-2.519
.326	-.186	.325	.079	-.438		.270	-1.396	-1.797	-2.049	-2.107	-2.090
.371	-.340	.411	-.296	-.491		.400	-1.027	-1.366	-1.631	-1.765	-1.942
.392	-.417	.471	-.869	-.597		.620	-1.067	-1.731	-1.870	-2.450	-2.679
.413	-.494	.531	-1.159	.186		.685	-2.740	-3.694	-3.077	-6.862	-14.423
.434	-.551	.564	-1.772	.623		.693	-2.516	-4.2[illegible]4	-4.615	-7.561	-13.417
.457	-.519	.559	-1.403	.696		.700	-1.521	-3.1[illegible]0	-3.634	-4.788	-10.353
.480	-.506	.553	-1.120	.656		.720	-.705	-1.4[illegible]2	-1.684	-2.358	-4.827
.502	-.526	.548	-1.041	.637		.750	-.533	-.8[illegible]2	-1.034	-1.515	-3.385
.551	-.417	.538	-.988	.597		.800	-.481	-.4[illegible]8	-.656	-.948	-2.494
.585	-.391	.531	-1.073	.696		.900	-.362	-.2[illegible]2	-.471	-.540	-1.519
.592	-.365	.504	-1.139	-1.114		.980	-.342	-.0[illegible]9	-.312	-.448	-.513
.613	-.250	.398	-.863	-.600							
.634	-.224	.292	-.573	-.338	Lower	.025	.711	.8[illegible]6	.822	.771	.577
.655	-.186	.159	-.342	-.358		.120	.883	.8[illegible]5	.763	.711	.455
.675	-.103	.027	-.204	-.252		.220	.836	.8[illegible]2	.796	.751	.532
.696	-.071	.040	-.092	-.093		.300	.738	.7[illegible]9	.729	.692	.449
.774	.006	.060	-.073	-.047		.620	.790	.8[illegible]6	.776	.685	-.103
.852	-.083	.080	-.053	[illegible]		.750	.856	.8[illegible]5	.729	.672	.462
.930	-.032	.106	-.026	.093		.850	.672	.723	.663	.612	.423
						.950	.454	.504	.391	.389	.295
α = 22.9°											
.032	-.237	.796	-.414	.047	Upper	.010	-9.019	-4.0[illegible]8	-4.384	-3.878	-3.899
.053	-.356	.577	-.529	-.094		.080	-2.122	-3.7[illegible]3	-4.492	-3.910	-2.555
.100	-.198	.398	-.579	-.363		.130	-3.455	-3.0[illegible]0	-3.490	-3.286	-2.661
.145	-.178	.279	-.541	-.424		.145	-10.013	-8.3[illegible]6	-7.228	-6.349	-7.113
.189	-.086	.312	-.478	-.350		.155	-4.271	-4.4[illegible]3	-4.297	-3.572	-3.188
.234	-.066	.345	-.159	-.424		.180	-2.792	-3.1[illegible]0	-3.194	-2.636	-2.496
.280	-.119	.371	.134	-.471		.220	-1.757	-2.2[illegible]8	-2.333	-1.955	-1.660
.326	-.211	.385	.038	-.585		.270	-1.485	-1.7[illegible]0	-1.869	-1.560	-1.284
.371	-.421	.491	-.433	-.686		.400	-1.081	-1.3[illegible]6	-1.446	-1.210	-1.218
.392	-.516	.551	-.949	-.888		.620	-1.015	-1.4[illegible]9	-1.358	-1.012	-1.133
.413	-.612	.610	-1.267	.182		.685	-3.826	-2.8[illegible]2	-.834	-1.879	-2.285
.434	-.639	.623	-1.993	.679		.693	-3.780	-3.2[illegible]6	-2.185	-2.070	-1.923
.457	-.580	.611	-1.509	.753		.700	-2.467	-2.5[illegible]3	-1.836	-1.452	-1.673
.480	-.501	.598	-1.299	.719		.720	-1.253	-1.2[illegible]3	-1.103	-1.038	-1.172
.502	-.501	.586	-1.140	.693		.750	-.869	-.8[illegible]6	-1.029	-1.025	-1.159
.551	-.329	.562	-1.038	.632		.800	-.577	-.4[illegible]7	-.948	-.968	-1.139
.585	-.277	.544	-1.153	.679		.900	-.405	-.3[illegible]8	-.881	-.891	-1.054
.592	-.263	.504	-1.439	-1.499		.980	-.405	-.2[illegible]6	-.780	-.898	-.994
.613	-.171	.385	-1.127	-.600							
.634	-.145	.279	-.681	-.471	Lower	.025	.829	.9[illegible]2	.861	.828	.665
.655	-.151	.159	-.376	-.350		.120	.935	.8[illegible]2	.800	.783	.573
.675	-.066	.027	-.197	-.121		.220	.882	.8[illegible]5	.841	.796	.652
.696	-.040	.053	-.096	.020		.300	.816	.7[illegible]6	.760	.739	.580
.774	-.040	.080	-.077	.030		.620	.809	.8[illegible]9	.773	.720	.250
.852	-.132	.106	-.057	.040		.750	.889	.8[illegible]5	.787	.720	.566
.930	-.053	.133	-.038	.148		.850	.696	.7[illegible]3	.625	.586	.454
						.950	.484	.5[illegible]4	.262	.236	.145

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TABLE 17 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 1.0 h_d/c = 0.5

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.6°											
.032	.257	.304	.282	.302		.010	.931	.875	.838	.815	.815
.053	.055	.074	.055	.074		.080	.414	.285	.247	.208	.288
.100	-.092	-.037	-.086	-.062		.130	-.426	-.695	-.820	-.790	-.735
.145	-.092	-.087	-.073	-.049		.145	-4.522	-4.535	-4.339	-4.832	-4.263
.189	-.012	-.031	-.018	-.018		.155	-1.680	-1.849	-1.861	-1.831	-1.421
.234	-.049	.031	-.012	.006		.180	-1.272	-1.203	-1.344	-1.225	-1.262
.280	-.049	.062	-.031	.031		.220	-.773	-.906	-.980	-1.060	-.827
.326	-.037	.087	-.024	.062		.270	-.621	-.831	-.888	-.790	-.661
.371	-.104	.136	-.098	.142	Upper	.400	-.657	-.831	-.918	-.876	-.698
.392	-.132	.149	-.006	.364		.620	-1.181	-1.303	-1.270	-1.549	-1.066
.413	-.159	.161	-.165	.025		.685	-6.353	-4.243	-3.513	-6.590	-5.561
.434	-.208	.180	-.410	-.136		.693	-6.268	-4.708	-4.999	-7.245	-4.569
.457	-.251	.200	-.551	-.092		.700	-3.937	-3.524	-3.803	-4.391	-3.424
.480	-.318	.220	-.502	.025		.720	-1.899	-1.669	-1.633	-1.917	-1.402
.502	-.392	.240	-.533	.129		.750	-1.260	-.893	-.949	-1.182	-1.084
.551	-.441	.280	-.704	.308		.800	-.828	-.434	-.573	-.668	-.943
.585	-.416	.310	-.864	.259		.900	-.517	-.329	-.382	-.367	-.778
.592	-.386	.310	-.998	-.906		.980	.012	-.174	-.197	-.080	-.600
.613	-.294	.248	-.815	-.863		.025	-.134	-.025	.111	.159	.037
.634	-.245	.186	-.582	-.820		.120	-.195	.000	.111	.116	.006
.655	-.196	.124	-.386	-.148		.220	-.152	-.031	.068	.092	.006
.675	-.135	.043	-.251	-.049	Lower	.300	.018	-.056	-.012	.043	-.031
.696	-.061	.019	-.171	-.006		.620	.511	.385	.160	.080	-.141
.774	-.049	.062	-.061	.025		.750	.797	.565	.253	.196	.227
.852	.000	-.025	-.012	-.117		.850	.718	.658	.431	.325	.398
.930	.073	-.199	.086	-.247		.950	.560	.496	.394	.367	.331
α = 5.6°											
.032	.096	.478	.142	.293		.010	.563	.510	.468	.523	.541
.053	-.102	.226	-.071	.081		.080	-.209	-.316	-.343	-.381	-.287
.100	-.185	.058	-.200	-.106		.130	-1.373	-1.671	-1.742	-1.768	-1.726
.145	-.146	-.019	-.194	-.094		.145	-6.807	-6.705	-6.150	-7.112	-6.482
.189	-.064	.039	-.123	-.044		.155	-2.809	-2.962	-2.891	-3.001	-2.560
.234	-.096	.110	-.032	-.037		.180	-2.050	-1.930	-2.029	-2.007	-2.057
.280	-.089	.123	.045	-.031		.220	-1.284	-1.420	-1.467	-1.646	-1.439
.326	-.096	.136	.006	-.031		.270	-.999	-1.220	-1.255	-1.258	-1.172
.371	-.191	.219	-.155	.000	Upper	.400	-.854	-1.065	-1.149	-1.207	-1.165
.392	-.245	.255	-.271	.131		.620	-1.145	-1.452	-1.399	-1.833	-1.713
.413	-.299	.290	-.587	.225		.685	-3.966	-4.240	-3.603	-7.228	-10.399
.434	-.376	.323	-.884	.212		.693	-3.524	-4.491	-4.976	-7.879	-9.533
.457	-.388	.340	-.807	.318		.700	-2.005	-3.343	-3.709	-4.724	-7.285
.480	-.439	.357	-.761	.393		.720	-.917	-1.581	-1.598	-2.026	-3.337
.502	-.522	.374	-.710	.425		.750	-.753	-.865	-.924	-1.323	-2.407
.551	-.503	.408	-.781	.456		.800	-.677	-.484	-.656	-.871	-1.917
.585	-.471	.432	-.832	.506		.900	-.550	-.432	-.606	-.729	-1.426
.592	-.446	.426	-.839	-.749		.980	-.449	-.342	-.500	-.613	-.618
.613	-.337	.336	-.671	-.640		.025	.183	.381	.406	.419	.223
.634	-.293	.232	-.497	-.531		.120	.234	.342	.362	.284	.096
.655	-.236	.136	-.342	-.393		.220	.538	.445	.431	.361	.248
.675	-.146	.026	-.200	-.131	Lower	.300	.620	.581	.574	.587	.516
.696	-.096	.032	-.097	.006		.620	.702	.716	.681	.691	.210
.774	-.006	.084	-.064	-.050		.750	.803	.761	.755	.736	.586
.852	-.038	.045	-.032	-.106		.850	.614	.626	.606	.613	.503
.930	.038	-.090	.058	-.125		.950	.386	.361	.306	.355	.299
α = 13.1°											
.032	-.077	.634	-.078	.247		.010	-.219	-1.387	-1.676	-1.780	-1.523
.053	-.245	.399	-.260	.007		.080	-1.013	-1.073	-1.242	-1.312	-1.162
.100	-.207	.209	-.370	-.187		.130	-2.472	-2.767	-3.125	-3.015	-3.039
.145	-.155	.105	-.312	-.187		.145	-8.680	-8.732	-8.587	-9.317	-8.609
.189	-.084	.137	-.266	-.160		.155	-3.672	-4.029	-4.314	-4.294	-3.678
.234	-.123	.196	-.097	-.200		.180	-2.510	-2.636	-2.971	-2.859	-2.827
.280	-.136	.203	.071	-.200		.220	-1.613	-1.877	-2.137	-2.196	-1.962
.326	-.155	.235	.045	-.240		.270	-1.207	-1.531	-1.736	-1.676	-1.575
.371	-.265	.314	-.247	-.240	Upper	.400	-1.020	-1.210	-1.436	-1.462	-1.452
.392	-.327	.376	-.604	-.234		.620	-1.110	-1.537	-1.656	-1.923	-2.001
.413	-.387	.438	-.988	.160		.685	-2.762	-4.010	-5.873	-6.438	-11.235
.434	-.439	.484	-1.293	.521		.693	-2.175	-3.892	-5.075	-6.192	-10.299
.457	-.458	.485	-1.078	.588		.700	-1.278	-2.998	-3.819	-3.391	-7.802
.480	-.484	.486	-.949	.541		.720	-.652	-1.361	-1.729	-1.364	-3.511
.502	-.536	.487	-.877	.507		.750	-.549	-.752	-1.062	-1.156	-2.485
.551	-.490	.489	-.903	.548		.800	-.490	-.458	-.708	-.871	-2.046
.585	-.445	.491	-.871	.608		.900	-.430	-.412	-.634	-.871	-1.749
.592	-.419	.477	-.845	-.795		.980	-.497	-.477	-.614	-.910	-1.213
.613	-.329	.379	-.689	-.500		.025	.497	.700	.721	.728	.568
.634	-.284	.255	-.520	-.394		.120	.813	.805	.761	.734	.542
.655	-.252	.124	-.344	.414		.220	.781	.765	.761	.728	.587
.675	-.155	.013	-.214	-.267	Lower	.300	.691	.700	.668	.650	.490
.696	-.110	.039	-.110	-.093		.620	.749	.791	.768	.695	.103
.774	-.013	.157	-.084	.060		.750	.952	.877	.801	.721	.542
.852	-.058	.078	-.058	-.060		.850	.632	.687	.601	.585	.439
.930	.000	.020	-.006	-.007		.950	.400	.412	.300	.292	.155

TABLE 17 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = .0 h_d/c = 0.5

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = 18.8°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.186	.754	-.282	.142
.053	-.321	.539	-.423	-.047
.100	-.218	.299	-.487	-.298
.145	-.192	.227	-.449	-.339
.189	-.090	.240	-.391	-.278
.234	-.096	.292	-.128	-.345
.280	-.147	.299	.122	-.359
.326	-.173	.331	.090	-.460
.371	-.340	.416	-.269	-.501
.392	-.410	.481	-.795	-.609
.413	-.481	.546	-1.090	.149
.434	-.538	.572	-1.679	.623
.457	-.519	.565	-1.321	.725
.480	-.513	.557	-1.058	.677
.502	-.526	.550	-.974	.637
.551	-.449	.536	-.936	.630
.585	-.410	.526	-1.038	.677
.592	-.378	.507	-1.122	-1.144
.613	-.276	.403	-.833	-.632
.634	-.224	.286	-.558	-.379
.655	-.205	.143	-.321	-.379
.675	-.122	.026	-.167	-.271
.696	-.071	.052	-.071	-.088
.774	-.006	.182	.064	-.051
.852	-.096	.078	-.064	-.014
.930	-.032	.104	-.019	.108

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-4.692	-3.057	-3.684	-3.506	-3.397
	.080	-1.756	-2.133	-3.386	-3.391	-2.051
	.130	-3.135	-3.015	-3.379	-3.218	-3.462
	.145	-9.712	-8.998	-8.789	-8.673	-9.333
	.155	-4.167	-4.483	-4.801	-4.500	-4.288
	.180	-2.782	-3.002	-3.440	-3.154	-3.346
	.220	-1.718	-2.150	-2.505	-2.391	-2.391
	.270	-1.288	-1.715	-2.025	-1.910	-1.974
	.400	-.974	-1.293	-1.571	-1.519	-1.808
	.620	-1.000	-1.507	-1.693	-1.756	-2.006
	.685	-2.891	-3.326	-3.115	-4.006	-9.987
	.693	-2.679	-3.489	-4.300	-4.038	-9.032
	.700	-1.622	-2.631	-3.304	-2.353	-6.795
	.720	-.756	-1.338	-1.557	-1.269	-3.064
	.750	-.551	-.806	-1.016	-1.083	-2.205
	.800	-.487	-.481	-.711	-.821	-1.821
	.900	-.385	-.312	-.569	-.718	-1.564
	.980	-.385	-.221	-.454	-.724	-1.115
Lower	.025	.705	.838	.819	.795	.615
	.120	.859	.838	.772	.712	.526
	.220	.821	.812	.826	.756	.583
	.300	.750	.730	.731	.686	.519
	.620	.769	.819	.785	.692	.096
	.750	.846	.877	.772	.744	.526
	.850	.679	.721	.630	.603	.436
	.950	.436	.437	.239	.233	.192

α = 22.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.231	.812	-.471	.025
.053	-.340	.637	-.577	-.101
.100	-.205	.396	-.656	-.354
.145	-.192	.305	-.610	-.392
.189	-.103	.325	-.550	-.354
.234	-.077	.357	-.186	-.405
.280	-.135	.377	.126	-.436
.326	-.212	.383	.106	-.538
.371	-.417	.500	-.431	-.614
.392	-.504	.552	-1.015	-.810
.413	-.590	.604	-1.313	.183
.434	-.603	.617	-2.095	.658
.457	-.558	.608	-1.578	.721
.480	-.506	.599	-1.346	.696
.502	-.506	.590	-1.174	.658
.551	-.346	.572	-1.094	.620
.585	-.314	.559	-1.174	.645
.592	-.282	.526	-1.505	-1.398
.613	-.192	.409	-1.207	-.770
.634	-.173	.279	-.729	-.455
.655	-.154	.149	-.431	-.367
.675	-.090	.019	-.232	-.152
.696	-.051	.039	-.113	.000
.774	-.013	.068	.020	.127
.852	-.128	.097	-.060	.019
.930	-.045	.136	-.040	.114

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-8.912	-3.950	-4.068	-4.032	-3.667
	.080	-2.109	-3.671	-4.175	-4.052	-2.346
	.130	-3.435	-2.995	-3.220	-3.408	-2.449
	.145	-10.080	-8.237	-6.927	-6.651	-6.545
	.155	-4.284	-4.418	-4.061	-3.773	-2.885
	.180	-2.832	-3.057	-2.998	-2.772	-2.244
	.220	-1.751	-2.215	-2.163	-2.056	-1.449
	.270	-1.446	-1.734	-1.733	-1.645	-1.109
	.400	-1.074	-1.233	-1.303	-1.280	-1.038
	.620	-1.008	-1.319	-1.221	-1.001	-.987
	.685	-3.873	-2.739	-.822	-2.062	-2.026
	.693	-3.820	-2.872	-2.024	-2.288	-1.635
	.700	-2.507	-2.215	-1.695	-1.538	-1.462
	.720	-1.280	-1.133	-1.025	-1.107	-1.013
	.750	-.855	-.730	-.924	-1.094	-1.019
	.800	-.603	-.526	-.860	-1.021	-1.006
	.900	-.451	-.330	-.784	-.968	-.949
	.980	-.424	-.273	-.671	-.935	-.878
Lower	.025	.809	.854	.822	.855	.660
	.120	.915	.854	.778	.789	.583
	.220	.889	.838	.810	.802	.654
	.300	.796	.773	.740	.736	.577
	.620	.802	.838	.759	.736	.263
	.750	.875	.877	.772	.763	.590
	.850	.690	.734	.595	.610	.487
	.950	.491	.458	.278	.259	.179

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TABLE 17 Continued
(b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.4°											
.032	-.231	.319	.288	.294	Upper	.010	.956	.877	.847	.851	.815
.053	.061	.096	.080	.060		.080	.517	.397	.349	.343	.365
.100	-.110	-.024	-.080	-.078		.130	-.246	-.469	-.553	-.533	-.584
.145	-.067	-.072	-.049	-.048		.145	-3.948	-3.798	-3.528	-4.085	-3.755
.189	.000	-.012	.006	-.006		.155	-1.400	-1.454	-1.406	-1.390	-1.156
.234	-.037	.054	-.018	.012		.180	-1.040	-.895	-1.004	-.913	-1.059
.280	-.037	.072	-.043	.042		.220	-.595	-.637	-.673	-.772	-.657
.326	-.012	.078	-.031	.078		.270	-.445	-.529	-.559	-.514	-.511
.371	-.085	.126	.049	.144		.400	-.439	-.439	-.457	-.508	-.499
.392	-.110	.135	.043	.331		.620	-.715	-.294	-.192	-.668	-.822
.413	-.134	.144	-.092	-.180		.685	-4.537	-.769	.024	-1.886	-6.007
.434	-.170	.156	-.312	-.300		.693	-4.435	-.919	-1.202	-1.990	-5.033
.457	-.201	.176	-.398	-.216		.700	-2.638	-.925	-1.052	-1.292	-3.511
.480	-.268	.196	-.361	-.048		.720	-1.064	-.739	-.709	-.753	-1.089
.502	-.341	.215	-.361	.096		.750	-.643	-.565	-.691	-.784	-.694
.551	-.377	.255	-.453	.288		.800	-.463	-.433	-.685	-.753	-.645
.585	-.371	.282	-.533	.270		.900	-.499	-.427	-.553	-.649	-.602
.592	-.353	.294	-.588	-.493		.980	-.397	-.409	-.559	-.551	-.450
.613	-.304	.240	-.453	-.559	Lower	.025	-.313	-.186	.006	.080	-.018
.634	-.256	.156	-.318	-.625		.120	-.367	-.144	-.018	.024	-.043
.655	-.231	.090	-.257	-.192		.220	-.313	-.174	-.018	-.012	-.043
.675	-.170	-.018	-.196	-.132		.300	-.060	-.222	-.090	-.049	-.055
.696	-.110	-.024	-.135	-.090		.620	.523	.397	.036	-.043	-.219
.774	.006	.048	-.012	-.099		.750	.787	.613	.102	.080	.043
.852	-.043	-.024	.018	-.108		.850	.679	.643	.294	.196	.189
.930	.049	-.132	.067	-.132		.950	.427	.361	.210	.190	.243
α = 5.8°											
.032	.076	.478	.137	.318	Upper	.010	.669	.647	.568	.624	.607
.053	-.114	.233	-.069	.100		.080	-.076	-.138	-.187	-.219	-.177
.100	-.190	.057	-.194	-.094		.130	-1.184	-1.364	-1.461	-1.442	-1.480
.145	-.152	-.006	-.162	-.075		.145	-6.298	-5.876	-5.420	-6.138	-5.795
.189	-.076	.057	-.106	-.025		.155	-2.547	-2.520	-2.460	-2.485	-2.176
.234	-.114	.126	-.050	-.025		.180	-1.828	-1.590	-1.680	-1.623	-1.765
.280	-.095	.126	.025	-.012		.220	-1.102	-1.125	-1.161	-1.286	-1.202
.326	-.089	.138	.031	[illegible]		.270	-.821	-.924	-.943	-.924	-.943
.371	-.177	.195	-.125	.050		.400	-.675	-.666	-.693	-.774	-.886
.392	-.225	.233	-.219	.200		.620	-.891	-.484	-.362	-.874	-1.234
.413	-.272	.270	-.487	.144		.685	-4.496	-1.163	-.150	-2.348	-8.565
.434	-.335	.308	-.762	.169		.693	-4.235	-.974	-1.317	-1.823	-7.743
.457	-.354	.322	-.712	.300		.700	-2.471	-.943	-1.136	-1.180	-5.744
.480	-.418	.337	-.599	.381		.720	-1.038	-.710	-.762	-.812	-2.334
.502	-.481	.351	-.549	.412		.750	-.694	-.654	-.706	-.799	-1.556
.551	-.455	.379	-.587	.450		.800	-.573	-.509	-.674	-.743	-1.227
.585	-.449	.402	-.649	.493		.900	-.624	-.528	-.543	-.674	-.974
.592	-.424	.396	-.674	-.643		.980	-.497	-.522	-.543	-.593	-.462
.613	-.354	.308	-.512	-.475	Lower	.025	.070	.339	.343	.337	.171
.634	-.316	.220	-.356	-.462		.120	.121	.270	.300	.250	.063
.655	-.291	.107	-.293	-.212		.220	.465	.289	.275	.250	.082
.675	-.196	-.031	-.237	-.144		.300	.611	.440	.412	.425	.418
.696	-.152	-.044	-.162	-.087		.620	.700	.685	.568	.674	.316
.774	.006	.082	-.006	-.090		.750	.828	.754	.737	.737	.607
.852	-.025	.019	.012	-.094		.850	.592	.616	.562	.574	.538
.930	.032	-.119	.062	-.106		.950	.357	.239	.212	.237	.329
α = 13.3°											
.032	-.077	.660	-.114	.277	Upper	.010	-.103	-1.000	-1.213	-1.170	-.452
.053	-.232	.436	-.278	.058		.080	-.878	-.910	-1.013	-1.056	-.903
.100	-.187	.218	-.361	-.174		.130	-2.259	-2.449	-2.652	-2.530	-2.626
.145	-.142	.128	-.316	-.181		.145	-8.254	-7.955	-7.497	-8.122	-7.770
.189	-.071	.173	-.247	-.129		.155	-3.420	-3.583	-3.646	-3.587	-3.194
.234	-.103	.231	-.101	-.161		.180	-2.323	-2.269	-2.472	-2.322	-2.433
.280	-.110	.237	.076	-.161		.220	-1.439	-1.596	-1.697	-1.733	-1.633
.326	-.136	.250	.063	-.226		.270	-1.071	-1.256	-1.316	-1.253	-1.278
.371	-.252	.333	-.202	-.213		.400	-.858	-.865	-.916	-.962	-1.129
.392	-.304	.385	-.538	-.161		.620	-1.052	-.744	-.658	-1.037	-1.484
.413	-.355	.436	-.873	.187		.685	-4.066	-2.077	-.839	-4.049	-9.286
.434	-.419	.474	-1.164	.510		.693	-3.401	-1.756	-1.375	-1.398	-8.396
.457	-.419	.472	-.949	.587		.700	-2.142	-1.122	-1.168	-.955	-6.202
.480	-.419	.470	-.810	.549		.720	-1.123	-.853	-.781	-.772	-2.530
.502	-.497	.468	-.734	.503		.750	-.839	-.750	-.736	-.753	-1.658
.551	-.426	.464	-.778	.549		.800	-.671	-.667	-.697	-.683	-1.310
.585	-.407	.462	-.810	.620		.900	-.574	-.609	-.561	-.595	-1.162
.592	-.394	.462	-.810	-.830		.980	-.413	-.647	-.574	-.658	-.736
.613	-.303	.372	-.645	-.581	Lower	.025	.471	.667	.652	.664	.549
.634	-.258	.256	-.468	-.458		.120	.800	.788	.749	.709	.613
.655	-.239	.128	-.342	-.148		.220	.761	.731	.736	.702	.639
.675	-.148	.032	-.228	-.065		.300	.678	.673	.665	.639	.536
.696	-.110	.038	-.133	-.006		.620	.736	.795	.768	.696	.239
.774	-.064	-.067	.038	.103		.750	.839	.872	.781	.734	.607
.852	-.019	.096	[illegible]	-.019		.850	.607	.641	.549	.569	.497
.930	.058	.006	.044	-.013		.950	.413	.263	.211	.247	.265

TABLE 17 Concluded
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = 19.0°$											
.032	-.172	.721	-.307	.169	Upper	.010	-3.486	-2.818	-2.898	-2.905	-2.977
.053	-.305	.494	-.454	-.026		.080	-1.569	-1.790	-2.423	-2.557	-1.691
.100	-.199	.321	-.534	-.273		.130	-2.948	-2.965	-2.976	-2.898	-3.263
.145	-.166	.200	-.481	-.286		.145	-9.230	-8.834	-7.900	-8.260	-8.919
.189	-.086	.240	-.414	-.240		.155	-3.954	-4.300	-4.126	-3.986	-3.992
.234	-.066	.274	-.160	-.318		.180	-2.606	-2.845	-2.846	-2.684	-3.084
.280	-.119	.294	.107	-.331		.220	-1.607	-1.997	-1.995	-1.923	-2.155
.326	-.139	.307	.000	-.390		.270	-1.189	-1.549	-1.527	-1.436	-1.757
.371	-.338	.421	-.287	-.435		.400	-.879	-1.075	-1.027	-1.022	-1.499
.392	-.401	.471	-.821	-.539		.620	-.924	-.795	-.708	-.955	-1.538
.413	-.464	.521	-1.122	.182		.685	-3.460	-1.322	-.435	-3.158	-7.991
.434	-.531	.548	-1.729	.611		.693	-3.441	-1.522	-1.663	-3.786	-7.089
.457	-.511	.545	-1.342	.689		.700	-2.214	-1.242	-1.449	-2.404	-5.272
.480	-.497	.542	-1.075	.656		.720	-1.151	-.988	-.994	-1.149	-2.182
.502	-.517	.539	-.962	.611		.750	-.854	-.888	-.897	-1.048	-1.492
.551	-.411	.533	-.962	.578		.800	-.645	-.788	-.754	-.982	-1.214
.585	-.365	.528	-1.062	.643		.900	-.493	-.735	-.663	-.908	-1.061
.592	-.325	.474	-1.269	-1.143		.980	-.399	-.748	-.669	-.835	-.716
.613	-.245	.361	-1.068	-.650	Lower	.025	.696	.828	.819	.815	.696
.634	-.206	.260	-.661	-.585		.120	.867	.848	.754	.741	.577
.655	-.186	.120	-.387	-.364		.220	.822	.795	.799	.748	.637
.675	-.113	.013	-.214	-.143		.300	.727	.748	.715	.708	.577
.696	-.060	.013	-.093	-.032		.620	.765	.835	.780	.721	.232
.774	.020	.040	.047	-.009		.750	.860	.875	.767	.748	.603
.852	-.046	.067	-.033	.013		.850	.652	.688	.572	.594	.511
.930	.013	.080	.013	.078		.950	.449	.287	.214	.254	.272
$\alpha = 23.1°$											
.032	-.240	.808	-.429	.052	Upper	.010	-8.307	-3.786	-3.840	-3.632	-3.459
.053	-.347	.608	-.526	-.097		.080	-1.982	-3.519	-3.917	-3.664	-1.970
.100	-.194	.401	-.585	-.355		.130	-3.231	-2.905	-3.052	-3.002	-2.404
.145	-.160	.294	-.559	-.381		.145	-9.517	-8.080	-6.550	-6.445	-6.377
.189	-.073	.321	-.487	-.336		.155	-4.003	-4.260	-3.775	-3.443	-2.711
.234	-.053	.361	-.169	-.413		.180	-2.623	-2.931	-2.762	-2.462	-2.077
.280	-.114	.374	.143	-.439		.220	-1.609	-2.070	-1.949	-1.774	-1.282
.326	-.200	.387	.091	-.549		.270	-1.334	-1.616	-1.529	-1.358	-.948
.371	-.414	.481	-.390	-.613		.400	-.988	-1.122	-1.065	-1.001	-.875
.392	-.501	.538	-.929	-.807		.620	-.929	-.868	-.903	-.838	-.861
.413	-.588	.594	-1.215	.187		.685	-3.800	-1.255	-.439	-1.696	-1.776
.434	-.628	.614	-1.923	.658		.693	-4.010	-1.282	-1.652	-1.910	-1.389
.457	-.568	.604	-1.403	.723		.700	-2.643	-1.162	-1.433	-1.260	-1.189
.480	-.521	.594	-1.163	.684		.720	-1.374	-.855	-.865	-.864	-.835
.502	-.487	.584	-1.027	.652		.750	-.955	-.795	-.800	-.871	-.848
.551	-.327	.564	-1.001	.600		.800	-.667	-.708	-.755	-.806	-.841
.585	-.280	.548	-1.052	.639		.900	-.471	-.641	-.671	-.825	-.815
.592	-.254	.514	-1.403	-1.349		.980	-.399	-.688	-.645	-.793	-.801
.613	-.167	.401	-1.312	-.850	Lower	.025	.798	.875	.820	.832	.721
.634	-.147	.274	-.793	-.574		.120	.909	.841	.761	.747	.641
.655	-.140	.127	-.455	-.290		.220	.850	.828	.787	.773	.694
.675	-.073	.013	-.273	-.110		.300	.791	.801	.716	.741	.648
.696	-.040	.040	-.156	-.026		.620	.778	.855	.768	.734	.327
.774	-.027	.077	.019	.103		.750	.877	.908	.800	.767	.628
.852	-.100	.114	-.065	.006		.850	.700	.701	.574	.617	.534
.930	-.027	.147	-.032	.129		.950	.510	.361	.265	.266	.214

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TABLE 18 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.7$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = -1.5°$											
.032	.262	.321	.284	.288	Upper	.010	.943	.905	.847	.841	.815
.053	.037	.075	.067	.054		.080	.438	.365	.282	.272	.268
.100	-.091	-.057	-.097	-.084		.130	-.396	-.622	-.721	-.683	-.706
.145	-.097	-.075	-.067	-.054		.145	-4.449	-4.355	-4.044	-4.487	-4.126
.189	-.024	-.019	.006	-.006		.155	-1.643	-1.735	-1.677	-1.621	-1.266
.234	-.061	.050	.018	.000		.180	-1.235	-1.131	-1.184	-1.082	-1.175
.280	-.061	.075	-.024	.024		.220	-.755	-.855	-.865	-.907	-.767
.326	-.061	.075	-.024	.072		.270	-.609	-.760	-.745	-.653	-.602
.371	-.122	.126	-.109	.138		.400	-.645	-.767	-.733	-.647	-.572
.392	-.037	.143	.000	.325		.620	-1.156	-1.301	-.673	-.623	-.633
.413	-.164	.163	-.151	-.108		.685	-6.353	-5.933	-1.028	-2.286	-3.092
.434	-.225	.163	-.381	-.252		.693	-6.287	-6.511	-2.121	-2.280	-2.714
.457	-.250	.190	-.496	-.162		.700	-3.968	-4.625	-1.677	-1.530	-2.331
.480	-.316	.210	-.466	.006		.720	-1.905	-1.911	-.974	-.974	-1.351
.502	-.396	.240	-.508	.114		.750	-1.266	-1.131	-.631	-.792	-.925
.551	-.420	.270	-.689	.300		.800	-.864	-.691	-.535	-.768	-.797
.585	-.420	.308	-.847	.258		.900	-.560	-.415	-.541	-.683	-.724
.592	-.408	.321	-.974	-.925		.980	.006	-.138	-.511	-.544	-.572
.613	-.310	.245	-.804	-.727	Lower	.025	-.146	-.132	.060	.139	-.006
.634	-.262	.176	-.587	-.799		.120	-.231	-.113	.048	.079	-.030
.655	-.213	.126	-.399	-.186		.220	-.189	-.126	.030	.054	
.675	-.116	.038	-.266	-.078		.300	-.030	-.189	-.036	.006	-.049
.696	-.073	.013	-.187	-.048		.620	.463	.377	.084	.006	-.146
.774	-.043	.063	-.073	.030		.750	.742	.616	.132	.115	.122
.852	-.018	-.031	.018	-.114		.850	.742	.716	.331	.218	.237
.930	.067	-.189	.085	-.246		.950	.572	.509	.252	.169	.164
$\alpha = 5.6°$											
.032	.074	.490	.147	.321	Upper	.010	.559	.546	.459	.500	.546
.053	-.105	.267	-.064	.094		.080	-.233	-.254	-.377	-.397	-.267
.100	-.192	.081	-.224	-.113		.130	-1.414	-1.557	-1.835	-1.756	-1.687
.145	-.155	.025	-.173	-.075		.145	-6.856	-6.414	-6.448	-7.058	-6.371
.189	-.068	.050	-.090	-.013		.155	-2.822	-2.810	-3.004	-2.968	-2.376
.234	-.105	.124	-.019	-.038		.180	-2.055	-1.824	-2.124	-1.974	-1.979
.280	-.105	.136	.045	-.038		.220	-1.301	-1.321	-1.533	-1.609	-1.383
.326	-.118	.149	.019	-.031		.270	-1.012	-1.154	-1.320	-1.224	-1.110
.371	-.199	.223	-.186	.000		.400	-.899	-1.036	-1.232	-1.160	-1.061
.392	-.260	.275	-.269	.163		.620	-1.112	-1.495	-1.483	-1.468	-1.309
.413	-.304	.329	-.596	.245		.685	-3.771	-6.359	-2.514	-3.872	-5.664
.434	-.360	.354	-.897	.245		.693	-3.306	-6.873	-3.570	-3.276	-4.833
.457	-.409	.370	-.840	.377		.700	-1.885	-4.950	-2.998	-2.468	-4.026
.480	-.459	.385	-.737	.446		.720	-.892	-2.103	-1.571	-1.212	-2.351
.502	-.540	.405	-.724	.452		.750	-.773	-1.253	-1.043	-.776	-1.706
.551	-.515	.425	-.788	.490		.800	-.666	-.713	-.767	-.724	-1.241
.585	-.490	.434	-.859	.540		.900	-.584	-.230	-.528	-.744	-.999
.592	-.471	.434	-.833	-.848		.980	-.478	.149	-.415	-.699	-.943
.613	-.329	.329	-.667	-.560	Lower	.025	.157	.416	.390	.417	.199
.634	-.310	.223	-.506	-.553		.120	.245	.366	.339	.276	.074
.655	-.267	.136	-.365	-.478		.220	.522	.465	.465	.353	.254
.675	-.155	.012	-.224	-.176		.300	.610	.596	.622	.564	.515
.696	-.093	.031	-.135	-.031		.620	.698	.720	.691	.654	.199
.774	-.043	.105	.000	.031		.750	.811	.782	.704	.692	.558
.852	-.062	.050	-.019	-.126		.850	.591	.676	.597	.551	.459
.930	.006	-.105	.045	-.132		.950	.358	.527	.308	.218	.143
$\alpha = 13.1°$											
.032	-.085	.620	-.078	.244	Upper	.010	-.250	-1.309	-1.583	-1.689	-1.387
.053	-.262	.430	-.247	.038		.080	-1.038	-1.018	-1.205	-1.267	-1.099
.100	-.222	.215	-.351	-.192		.130	-2.462	-2.638	-2.968	-2.924	-2.937
.145	-.157	.127	-.331	-.186		.145	-8.660	-8.439	-8.205	-9.044	-8.399
.189	-.105	.158	-.260	-.128		.155	-3.654	-3.897	-4.103	-4.165	-3.467
.234	-.105	.215	-.052	-.173		.180	-2.500	-2.524	-2.795	-2.709	-2.688
.280	-.144	.209	.078	-.173		.220	-1.596	-1.809	-2.000	-2.066	-1.832
.326	-.157	.221	.026	-.218		.270	-1.237	-1.480	-1.635	-1.566	-1.452
.371	-.288	.335	-.273	-.237		.400	-1.038	-1.177	-1.314	-1.299	-1.282
.392	-.360	.385	-.611	-.205		.620	-1.045	-1.417	-1.135	-1.202	-1.079
.413	-.406	.430	-.975	.179		.685	-2.558	-5.617	-1.660	-3.417	-4.042
.434	-.477	.455	-1.293	.526		.693	-2.147	-6.067	-2.731	-2.937	-3.571
.457	-.477	.460	-1.072	.590		.700	-1.340	-4.321	-2.212	-1.949	-3.002
.480	-.497	.465	-.949	.551		.720	-.647	-1.797	-1.410	-1.273	-1.858
.502	-.563	.470	-.897	.526		.750	-.577	-1.050	-1.045	-1.098	-1.387
.551	-.510	.475	-.910	.564		.800	-.468	-.607	-.987	-1.078	-1.249
.585	-.477	.481	-.877	.609		.900	-.436	-.367	-1.026	-1.117	-1.171
.592	-.432	.468	-.884	-.833		.980	-.397	-.063	-.878	-.988	-1.119
.613	-.314	.367	-.715	-.450	Lower	.025	.494	.696	.692	.741	.576
.634	-.301	.266	-.513	-.404		.120	.821	.816	.737	.734	.576
.655	-.249	.127	-.344	-.429		.220	.782	.772	.731	.734	.595
.675	-.144	.019	-.208	-.269		.300	.654	.690	.641	.650	.504
.696	-.111	.038	-.110	-.071		.620	.744	.791	.712	.689	.209
.774	-.013	.152	.026	.077		.750	.840	.873	.744	.695	.576
.852	-.085	.076	-.045	-.058		.850	.609	.690	.558	.539	.438
.930	.000	.038	.013	.000		.950	.378	.468	.179	.143	.072

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TABLE 18 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 1.7 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 18.8°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.174	.761	-.295	.171
.053	-.334	.568	-.436	-.026
.100	-.227	.323	-.513	-.290
.145	-.180	.207	-.468	-.323
.189	-.107	.239	-.410	-.257
.234	-.107	.284	-.115	-.316
.280	-.160	.284	.096	-.342
.326	-.187	.303	.096	-.435
.371	-.374	.413	-.333	-.481
.392	-.450	.470	-.846	-.599
.413	-.521	.529	-1.096	.178
.434	-.588	.549	-1.718	.626
.457	-.548	.548	-1.359	.698
.480	-.534	.546	-1.096	.659
.502	-.568	.544	-1.013	.645
.551	-.494	.543	-.955	.599
.585	-.441	.542	-1.051	.659
.592	-.407	.516	-1.160	-1.232
.613	-.267	.413	-.885	-.750
.634	-.287	.271	-.596	-.421
.655	-.240	.142	-.372	-.408
.675	-.134	.019	-.224	-.263
.696	-.107	.039	-.096	-.066
.774	-.013	.207	.032	.105
.852	-.107	.077	-.064	-.026
.930	-.040	.116	-.038	.105

Surface	x/c	Wing, flap, or aileron 0.221	0.421	0.640	0.800	0.918
Upper	.010	-4.448	-3.02	-3.418	-3.256	-3.152
	.080	-1.786	-2.20	-3.141	-3.154	-1.563
	.130	-3.192	-2.96	-3.174	-3.090	-3.072
	.145	-9.929	-8.81	-8.384	-8.103	-8.240
	.155	-4.265	-4.41	-4.518	-4.173	-3.519
	.180	-2.826	-2.949	-3.201	-2.897	-2.731
	.220	-1.760	-2.130	-2.298	-2.141	-1.843
	.270	-1.334	-1.68	-1.851	-1.692	-1.456
	.400	-1.007	-1.30	-1.409	-1.295	-1.536
	.620	-.968	-1.48	-1.238	-1.147	-1.302
	.685	-2.970	-4.569	-1.271	-2.385	-2.157
	.693	-2.747	-5.01	-2.489	-2.590	-1.923
	.700	-1.701	-3.56	-2.088	-2.000	-1.709
	.720	-.805	-1.53	-1.449	-1.513	-1.242
	.750	-.615	-.68	-1.153	-1.192	-1.229
	.800	-.536	-.51	-.948	-1.109	-1.215
	.900	-.432	-.31	-.909	-1.000	-1.162
	.980	-.347	-.10	-.869	-.955	-1.095
Lower	.025	.693	.85	.830	.801	.674
	.120	.903	.85	.771	.718	.594
	.220	.850	.82	.810	.731	.648
	.300	.746	.74	.724	.679	.588
	.620	.805	.839	.764	.667	.247
	.750	.863	.865	.777	.705	.581
	.850	.674	.716	.586	.538	.447
	.950	.438	.490	.224	.179	.093

α = 22.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.240	.835	-.487	.052
.053	-.338	.641	-.586	-.118
.100	-.201	.407	-.665	-.373
.145	-.182	.294	-.599	-.419
.189	-.091	.314	-.540	-.347
.234	-.065	.347	-.138	-.412
.280	-.117	.374	.119	-.451
.326	-.208	.387	.059	-.543
.371	-.416	.507	-.474	-.634
.392	-.516	.560	-1.041	-.831
.413	-.617	.614	-1.317	.203
.434	-.637	.634	-2.042	.674
.457	-.572	.610	-1.614	.739
.480	-.533	.127	-1.376	.693
.502	-.500	.565	-1.225	.674
.551	-.331	.550	-1.106	.621
.585	-.286	.541	-1.232	.680
.592	-.266	.521	-1.587	-1.524
.613	-.149	.427	-1.264	-.900
.634	-.169	.294	-.764	-.484
.655	-.156	.160	-.435	-.360
.675	-.065	.027	-.237	-.111
.696	-.045	.053	-.112	.033
.774	-.019	.214	.026	.137
.852	-.110	.107	-.092	.039
.930	-.039	.147	-.040	.144

Surface	x/c	Wing, flap, or aileron 0.221	0.421	0.640	0.800	0.918
Upper	.010	-9.365	-4.14	-4.258	-4.057	-3.794
	.080	-2.107	-3.95	-4.369	-4.096	-2.469
	.130	-3.431	-3.05	-3.369	-3.451	-2.540
	.145	-9.866	-8.23	-7.149	-6.797	-6.783
	.155	-4.207	-4.48	-4.206	-3.833	-2.943
	.180	-2.740	-3.09	-3.107	-2.819	-2.345
	.220	-1.739	-2.27	-2.257	-2.101	-1.546
	.270	-1.462	-1.80	-1.799	-1.673	-1.169
	.400	-1.106	-1.36	-1.380	-1.324	-1.130
	.670	-1.014	-1.43	-1.289	-1.027	-1.065
	.685	-3.991	-3.60	-.648	-1.811	-1.852
	.693	-3.912	-3.94	-1.943	-1.956	-1.598
	.700	-2.582	-2.84	-1.694	-1.502	-1.436
	.720	-1.330	-1.25	-1.125	-1.139	-1.027
	.750	-.896	-.81	-1.040	-1.093	-1.052
	.800	-.612	-.56	-.988	-1.047	-1.065
	.900	-.461	-.40	-.935	-1.001	-.994
	.980	-.402	-.16	-.831	-.962	-.929
Lower	.025	.823	.88	.844	.803	.689
	.120	.929	.88	.778	.738	.598
	.220	.883	.86	.844	.777	.663
	.300	.803	.78	.759	.738	.611
	.620	.810	.84	.798	.692	.266
	.750	.883	.88	.765	.724	.598
	.850	.711	.74	.608	.580	.481
	.950	.474	.50	.249	.211	.182

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TABLE 18 (b) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = −1.8°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.276	.285	.277	.290
.053	.067	.062	.074	.055
.100	-.073	-.031	-.092	-.080
.145	-.080	-.081	-.055	-.031
.189	-.006	-.019	-.012	.000
.234	-.024	.062	.018	.031
.280	-.043	.074	-.025	.049
.326	-.031	.081	-.018	.092
.371	-.092	.143	-.068	.154
.392	-.012	.155	-.012	.376
.413	-.165	.167	-.173	.105
.434	-.202	.192	-.431	-.037
.457	-.239	.210	-.536	-.012
.480	-.300	.230	-.505	.086
.502	-.386	.250	-.549	.160
.551	-.429	.270	-.740	.265
.585	-.423	.292	-.906	.240
.592	-.398	.292	-1.054	-.943
.613	-.288	.223	-.900	-.696
.634	-.245	.180	-.641	-.820
.655	-.190	.130	-.431	-.166
.675	-.104	.037	-.290	-.062
.696	-.043	.037	-.185	-.025
.774	-.031	.068	-.074	.018
.852	.006	-.019	.006	-.129
.930	.098	-.199	.092	-.259

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.906	.825	.814	.851	.802
	.080	.398	.248	.216	.240	.257
	.130	-.478	-.782	-.875	-.777	-.778
	.145	-4.630	-4.671	-4.407	-4.869	-4.379
	.155	-1.745	-1.898	-1.868	-1.794	-1.390
	.180	-1.317	-1.259	-1.362	-1.227	-1.280
	.220	-.815	-.962	-1.005	-1.060	-.882
	.270	-.655	-.875	-.937	-.814	-.692
	.400	-.710	-.912	-1.005	-.918	-.723
	.620	-1.231	-1.551	-1.492	-1.522	-1.109
	.685	-6.535	-6.743	-4.087	-5.344	-4.795
	.693	-6.461	-7.339	-6.034	-6.509	-4.397
	.700	-4.103	-5.285	-4.789	-4.364	-3.497
	.720	-1.966	-2.301	-2.219	-2.219	-1.500
	.750	-1.305	-1.396	-1.362	-1.393	-1.200
	.800	-.913	-.813	-.752	-.795	-1.035
	.900	-.619	-.217	-.173	-.327	-.839
	.980	-.037	.236	.142	-.012	-.631
Lower	.025	-.043	.050	.179	.210	.043
	.120	-.110	.074	.142	.136	-.006
	.220	-.073	.043	.111	.136	-.006
	.300	.043	.006	.043	.049	-.043
	.620	.398	.366	.222	.111	-.018
	.750	.625	.509	.253	.228	.251
	.850	.710	.664	.499	.333	.343
	.950	.563	.602	.462	.339	.227

α = 5.5°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.062	.474	.125	.301
.053	-.119	.253	-.081	.083
.100	-.206	.076	-.212	-.115
.145	-.150	-.006	-.181	-.103
.189	-.069	.057	-.125	-.051
.234	-.112	.120	-.037	-.051
.280	-.106	.133	.031	-.064
.326	-.112	.145	-.019	-.032
.371	-.225	.215	-.206	-.006
.392	-.270	.250	-.293	.122
.413	-.312	.297	-.593	.244
.434	-.368	.335	-.912	.269
.457	-.400	.345	-.862	.391
.480	-.450	.365	-.768	.462
.502	-.537	.385	-.737	.455
.551	-.531	.405	-.824	.500
.585	-.487	.418	-.874	.538
.592	-.456	.411	-.830	-.865
.613	-.300	.316	-.674	-.550
.634	-.318	.209	-.531	-.571
.655	-.262	.101	-.393	-.506
.675	-.150	.006	-.250	-.212
.696	-.087	.019	-.150	-.058
.774	-.019	.101	.006	[illegible]
.852	-.056	.032	-.019	-.141
.930	.037	-.101	.044	-.141

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.535	.487	.391	.381	.431
	.080	-.229	-.316	-.462	-.512	-.387
	.130	-1.426	-1.664	-1.981	-1.936	-1.911
	.145	-6.928	-6.693	-6.814	-7.443	-6.918
	.155	-2.866	-2.942	-3.205	-3.184	-2.660
	.180	-2.095	-1.917	-2.282	-2.135	-2.242
	.220	-1.337	-1.423	-1.667	-1.754	-1.573
	.270	-1.044	-1.246	-1.462	-1.374	-1.311
	.400	-.930	-1.132	-1.410	-1.374	-1.299
	.620	-1.127	-1.651	-1.872	-2.110	-1.917
	.685	-3.649	-6.674	-4.731	-7.043	-9.041
	.693	-3.222	-7.136	-6.994	-8.510	-9.122
	.700	-1.859	-5.219	-5.577	-5.994	-7.418
	.720	-.860	-2.246	-2.615	-3.153	-3.834
	.750	-.751	-1.360	-1.654	-2.079	-2.803
	.800	-.675	-.765	-.974	-1.361	-2.185
	.900	-.573	-.240	-.301	-.562	-1.386
	.980	-.458	.158	.064	-.037	-.537
Lower	.025	.191	.424	.308	.493	.100
	.120	.267	.405	.391	.425	.262
	.220	.535	.500	.692	.500	.537
	.300	.592	.595	.641	.543	.450
	.620	.681	.727	.692	.662	-.044
	.750	.790	.784	.692	.662	.487
	.850	.605	.677	.615	.587	.450
	.950	.376	.538	.449	.406	.244

α = 18.6°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.183	.728	-.329	.151
.053	-.314	.541	-.474	-.027
.100	-.196	.327	-.520	-.282
.145	-.183	.227	-.487	-.337
.189	-.092	.247	-.448	-.275
.234	-.097	.287	-.132	-.323
.280	-.137	.307	.112	-.371
.326	-.196	.327	.046	-.446
.371	-.340	.434	-.310	-.515
.392	-.422	.485	-.922	-.632
.413	-.504	.534	-1.159	.179
.434	-.563	.568	-1.818	.646
.457	-.549	.565	-1.427	.728
.480	-.504	.560	-1.153	.701
.502	-.549	.555	-1.087	.666
.551	-.432	.550	-1.034	.625
.585	-.392	.541	-1.172	.694
.592	-.366	.528	-1.159	-1.174
.613	-.235	.414	-.850	-.750
.634	-.216	.300	-.573	-.343
.655	-.183	.180	-.356	-.357
.675	-.105	.047	-.211	-.261
.696	-.065	.053	-.099	-.103
.774	.007	.194	.026	.124
.852	-.052	.093	-.053	.007
.930	-.013	.120	-.046	.103

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-5.349	-3.359	-4.073	-4.004	-3.853
	.080	-1.936	-2.531	-3.922	-4.017	-2.695
	.130	-3.359	-3.105	-3.482	-3.556	-3.689
	.145	-10.170	-9.261	-9.018	-9.016	-9.792
	.155	-4.414	-4.654	-5.021	-4.847	-4.579
	.180	-2.985	-3.145	-3.666	-3.491	-3.650
	.220	-1.850	-2.304	-2.706	-2.680	-2.636
	.270	-1.436	-1.836	-2.212	-2.180	-2.185
	.400	-1.075	-1.416	-1.786	-1.837	-2.047
	.620	-1.008	-1.696	-2.109	-2.476	-2.518
	.685	-2.798	-4.681	-4.032	-6.441	-11.133
	.693	-2.644	-5.108	-6.023	-7.442	-10.734
	.700	-1.636	-3.599	-4.787	-5.150	-8.582
	.720	-.748	-1.516	-2.301	-2.839	-4.585
	.750	-.608	-.821	-1.442	-1.903	-3.284
	.800	-.514	-.414	-.817	-1.179	-2.433
	.900	-.381	-.214	-.275	-.514	-1.550
	.980	-.307	-.053	-.007	-.356	-1.105
Lower	.025	.728	.841	.831	.777	.582
	.120	.895	.841	.790	.705	.451
	.220	.868	.828	.838	.738	.536
	.300	.768	.761	.776	.685	.451
	.620	.808	.835	.810	.659	-.111
	.750	.908	.868	.749	.652	.451
	.850	.668	.741	.728	.586	.399
	.950	.454	.541	.508	.356	.164

TABLE 18 Continued (b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 1.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 22.9°

x/l	Fuselage, 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.257	.807	-.445	.041
.053	-.362	.632	-.556	-.130
.100	-.224	.407	-.608	-.382
.145	-.184	.303	-.569	-.423
.189	-.092	.323	-.517	-.375
.234	-.079	.348	-.137	-.436
.280	-.145	.374	.131	-.477
.326	-.224	.387	.092	-.607
.371	-.435	.490	-.484	-.689
.392	-.510	.535	-1.014	-.927
.413	-.619	.587	-1.289	.184
.434	-.632	.620	-2.041	.689
.457	-.593	.600	-1.589	.743
.480	-.540	.580	-1.341	.709
.502	-.520	.565	-1.190	.675
.551	-.342	.555	-1.092	.634
.585	-.303	.542	-1.236	.668
.592	-.263	.510	-1.570	-1.555
.613	-.145	.407	-1.223	-.900
.634	-.178	.284	-.726	-.498
.655	-.158	.148	-.406	-.348
.675	-.092	.032	-.216	-.102
.696	-.066	.058	-.098	.020
.774	-.053	.213	.033	.130
.852	-.138	.110	-.065	.020
.930	-.053	.136	-.033	.130

Surface	x/c	Wing, flap, or aileron, 0.221	0.426	0.640	0.800	0.918
Upper	.010	-9.141	-[illegible].975	-4.501	-4.114	-3.912
	.080	-2.134	-[illegible].762	-4.603	-4.127	-2.687
	.130	-3.431	-2.981	-3.601	-3.539	-2.615
	.145	-9.984	-[illegible].015	-7.235	-6.332	-7.001
	.155	-4.241	-4.317	-4.330	-3.670	-3.016
	.180	-2.792	-3.014	-3.239	-2.767	-2.476
	.220	-1.745	-2.201	-2.387	-2.087	-1.627
	.270	-1.469	-1.749	-1.930	-1.707	-1.251
	.400	-1.106	-1.304	-1.487	-1.374	-1.212
	.620	-1.027	-[illegible].407	-1.391	-1.171	-1.120
	.685	-3.932	-[illegible].640	-.730	-1.943	-2.391
	.693	-3.840	-[illegible].053	-2.148	-2.139	-2.075
	.700	-2.542	-[illegible].923	-1.841	-1.563	-1.831
	.720	-1.304	-[illegible].265	-1.146	-1.158	-1.238
	.750	-.876	.781	-1.057	-1.125	-1.192
	.800	-.586	.484	-1.016	-1.066	-1.166
	.900	-.421	.303	-.934	-.975	-1.067
	.980	-.356	.058	-.825	-.968	-.988
Lower	.025	.797	.858	.866	.837	.652
	.120	.909	.858	.805	.765	.586
	.220	.876	.845	.832	.798	.639
	.300	.784	.794	.791	.739	.586
	.620	.790	.832	.798	.706	.237
	.750	.869	.865	.798	.726	.573
	.850	.692	.742	.627	.582	.461
	.950	.474	.516	.259	.222	.151

L-927

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TABLE 18 Continued
(c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 4.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing , flap , or aileron				
α = -1.4°											
.032	.269	.302	.285	.282	Upper	.010	.980	.883	.839	.831	.839
.053	.061	.054	.062	.049		.080	.505	.393	.343	.335	.374
.100	-.098	-.042	-.118	-.092		.130	-.302	-.508	-.576	-.515	-.459
.145	-.080	-.085	-.074	-.049		.145	-4.278	-3.985	-3.620	-3.964	-3.454
.189	-.012	-.030	-.012	-.012		.155	-1.553	-1.548	-1.445	-1.321	-.962
.234	-.049	.054	-.006	.012		.180	-1.159	-.986	-.992	-.831	-.894
.280	-.024	.067	-.037	.024		.220	-.703	-.720	-.668	-.689	-.533
.326	-.031	.085	-.037	.086		.270	-.542	-.629	-.557	-.422	-.386
.371	-.080	.121	-.099	.135		.400	-.579	-.587	-.404	-.292	-.251
.392	-.031	.135	.012	.331		.620	-1.060	-.931	.159	.180	.257
.413	-.141	.145	-.124	-.245		.685	-6.022	-4.396	.122	-1.117	-1.678
.434	-.178	.151	-.341	-.343		.693	-5.954	-4.832	-1.090	-1.303	-1.396
.457	-.214	.175	-.447	-.165		.700	-3.735	-3.387	-.968	-.966	-1.256
.480	-.276	.195	-.409	-.018		.720	-1.781	-1.252	-.661	-.763	-.931
.502	-.349	.215	-.434	.122		.750	-1.183	-.726	-.668	-.800	-.937
.551	-.398	.235	-.608	.331		.800	-.777	-.568	-.661	-.831	-.937
.585	-.392	.266	-.738	.282		.900	-.512	-.581	-.619	-.751	-.747
.592	-.374	.266	-.831	-.845		.980	-.037	-.496	-.588	-.720	-.668
.613	-.282	.224	-.682	-.649							
.634	-.251	.151	-.478	-.674	Lower	.025	-.351	-.194	-.012	.074	-.049
.655	-.202	.103	-.329	-.159		.120	-.376	-.145	-.012	.025	-.049
.675	-.135	.024	-.223	-.092		.220	-.314	-.163	-.037	.006	-.049
.696	-.092	.012	-.161	-.018		.300	-.074	-.230	-.098	-.062	-.067
.774	.006	.048	-.056	.024		.620	.542	.375	-.031	-.130	-.251
.852	-.006	-.036	.012	-.098		.750	.807	.605	.031	-.031	-.037
.930	.073	-.169	.087	-.208		.850	.703	.659	.306	.099	.031
						.950	.536	.369	.214	.050	
α = 5.8°											
.032	.070	.481	.132	.306	Upper	.010	.654	.667	.624	.660	.669
.053	-.102	.250	-.050	.096		.080	-.115	-.141	-.153	-.151	-.051
.100	-.191	.058	-.195	-.102		.130	-1.237	-1.391	-1.433	-1.307	-1.261
.145	-.153	.013	-.157	-.064		.145	-6.442	-6.109	-5.425	-5.744	-5.234
.189	-.057	.051	-.069	-.038		.155	-2.590	-2.615	-2.452	-2.256	-1.783
.234	-.089	.135	.019	-.032		.180	-1.904	-1.673	-1.668	-1.433	-1.496
.280	-.102	.122	.057	-.019		.220	-1.160	-1.186	-1.121	-1.081	-.942
.326	-.102	.135	.031	-.025		.270	-.897	-.987	-.879	-.723	-.662
.371	-.172	.199	-.163	.038		.400	-.769	-.814	-.592	-.452	-.465
.392	-.215	.205	-.239	.191		.620	-1.141	-1.006	.102	.151	.229
.413	-.261	.276	-.490	.140		.685	-5.724	-4.481	.096	-1.075	-1.707
.434	-.337	.321	-.786	.178		.693	-5.551	-4.897	-1.178	-1.301	-1.401
.457	-.369	.340	-.748	.331		.700	-3.436	-3.481	-1.070	-.955	-1.248
.480	-.420	.360	-.647	.414		.720	-1.654	-1.314	-.720	-.742	-.879
.502	-.490	.380	-.622	.433		.750	-1.167	-.840	-.720	-.786	-.891
.551	-.484	.405	-.735	.490		.800	-.872	-.788	-.739	-.830	-.891
.585	-.471	.436	-.855	.484		.900	-.564	-.795	-.720	-.817	-.917
.592	-.446	.429	-.943	-.923		.980	-.109	-.750	-.688	-.754	-.853
.613	-.312	.353	-.723	-.643							
.634	-.306	.244	-.490	-.522	Lower	.025	.026	.340	.331	.352	.166
.655	-.255	.128	-.333	-.121		.120	.115	.256	.293	.251	.089
.675	-.178	.006	-.233	-.051		.220	.551	.301	.274	.251	.089
.696	-.115	-.006	-.163	-.025		.300	.603	.423	.427	.365	.325
.774	-.006	.083	-.044	.051		.620	.699	.718	.650	.635	.376
.852	-.025	.038	.013	-.089		.750	.821	.782	.713	.679	.624
.930	.070	-.147	.088	-.153		.850	.615	.628	.535	.528	.503
						.950	.474	.231	.159	.138	.140
α = 13.3°											
.032	-.057	.635	-.071	.277	Upper	.010	-.110	-.987	-1.159	-.975	-.119
.053	-.239	.442	-.247	.046		.080	-.903	-.923	-.994	-.975	-.698
.100	-.195	.218	-.351	-.171		.130	-2.310	-2.462	-2.608	-2.345	-2.231
.145	-.138	.115	-.305	-.191		.145	-8.428	-8.051	-7.508	-7.770	-6.831
.189	-.088	.147	-.260	-.145		.155	-3.498	-3.641	-3.602	-3.326	-2.621
.234	-.088	.224	-.078	-.171		.180	-2.381	-2.314	-2.384	-2.098	-2.005
.280	-.119	.244	.065	-.165		.220	-1.497	-1.654	-1.640	-1.559	-1.282
.326	-.119	.231	.013	-.224		.270	-1.123	-1.314	-1.245	-1.072	-.911
.371	-.251	.333	-.273	-.224		.400	-.929	-.981	-.790	-.617	-.597
.392	-.300	.375	-.578	-.171		.620	-.974	-.955	-.026	.110	.201
.413	-.352	.423	-.916	.211		.685	-3.149	-4.179	-.007	-1.124	-1.659
.434	-.415	.474	-1.228	.527		.693	-2.801	-4.532	-1.278	-1.351	-1.351
.457	-.415	.480	-1.007	.619		.700	-1.775	-3.212	-1.159	-1.007	-1.213
.480	-.440	.485	-.858	.573		.720	-.962	-1.179	-.790	-.773	-.836
.502	-.515	.480	-.780	.527		.750	-.826	-.731	-.803	-.832	-.848
.551	-.465	.475	-.851	.566		.800	-.691	-.808	-.817	-.903	-.861
.585	-.421	.474	-.864	.619		.900	-.549	-.821	-.817	-.812	-.886
.592	-.406	.474	-.864	-.830		.980	-.426	-.795	-.784	-.767	-.823
.613	-.289	.372	-.715	-.640							
.634	-.283	.231	-.513	-.560	Lower	.025	.432	.667	.672	.669	.534
.655	-.258	.128	-.338	-.395		.120	.600	.827	.784	.728	.578
.675	-.163	-.006	-.201	-.165		.220	.768	.763	.790	.728	.628
.696	-.119	.013	-.110	-.040		.300	.658	.673	.665	.643	.528
.774	[illegible]	.141	.039	.072		.620	.729	.782	.751	.682	.321
.852	-.069	.058	-.013	-.072		.750	.832	.859	.790	.708	.635
.930	.019	-.006	.045	-.026		.850	.626	.641	.540	.539	.465
						.950	.387	.256	.158	.156	.138

TABLE 18 (c) Continued Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 4.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = 19.0°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.184	.777	-.294	.136
.053	-.316	.560	-.445	-.013
.100	-.211	.323	-.504	-.258
.145	-.171	.237	-.471	-.284
.189	-.092	.277	-.419	-.232
.234	-.105	.316	-.098	-.290
.280	-.132	.316	.124	-.310
.326	-.171	.323	.085	-.387
.371	-.336	.415	-.327	-.419
.392	-.400	.465	-.831	-.516
.413	-.481	.533	-1.099	.168
.434	-.547	.566	-1.688	.594
.457	-.514	.560	-1.347	.665
.480	-.494	.550	-1.060	.632
.502	-.527	.540	-.962	.600
.551	-.435	.530	-.962	.568
.585	-.395	.527	-1.027	.632
.592	-.369	.487	-1.190	-1.168
.613	-.231	.402	-.975	-.800
.634	-.244	.270	-.648	-.555
.655	-.217	.119	-.386	-.439
.675	-.138	[illegible]	-.216	-.194
.696	-.105	.040	-.111	-.032
.774	[illegible]	.184	.039	.103
.852	-.079	.119	-.046	-.039
.930	-.026	.105	-.007	.071

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-3.957	-2.733	-2.878	-2.924	-2.529
	.080	-1.683	-1.739	-2.465	-2.630	-1.153
	.130	-3.060	-2.931	-2.852	-2.754	-2.746
	.145	-9.596	-8.786	-7.615	-7.777	-7.383
	.155	-4.106	-4.294	-3.975	-3.768	-2.977
	.180	-2.709	-2.806	-2.749	-2.512	-2.292
	.220	-1.696	-1.996	-1.910	-1.805	-1.442
	.270	-1.247	-1.567	-1.446	-1.289	-1.080
	.400	-.955	-1.120	-.891	-.791	-1.159
	.620	-.981	-1.172	-.336	-.216	-.942
	.685	-3.320	-3.543	-.123	-1.151	-1.857
	.693	-3.157	-3.767	-1.349	-1.419	-1.554
	.700	-1.995	-2.694	-1.239	-1.073	-1.337
	.720	-1.027	-1.087	-.891	-.850	-.922
	.750	-.799	-.803	-.903	-.896	-.929
	.800	-.682	-.896	-.916	-.909	-.909
	.900	-.552	-.929	-.852	-.791	-.876
	.980	-.435	-.883	-.858	-.752	-.810
Lower	.025	.682	.843	.807	.824	.678
	.120	.877	.850	.749	.752	.593
	.220	.845	.803	.781	.785	.639
	.300	.728	.751	.710	.693	.586
	.620	.754	.803	.761	.713	.310
	.750	.825	.876	.755	.733	.606
	.850	.656	.652	.536	.595	.487
	.950	.409	.323	.168	.229	.178

α = 22.8°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.234	.842	-.449	.066
.053	-.341	.656	-.532	-.105
.100	-.207	.418	-.590	-.356
.145	-.180	.318	-.558	-.402
.189	-.060	.318	-.513	-.342
.234	-.060	.378	-.122	-.395
.280	-.100	.371	.128	-.448
.326	-.207	.405	.038	-.533
.371	-.421	.477	-.429	-.612
.392	-.500	.530	-.962	-.790
.413	-.588	.590	-1.224	.191
.434	-.621	.623	-1.936	.665
.457	-.566	.605	-1.468	.731
.480	-.514	.585	-1.224	.698
.502	-.487	.570	-1.077	.672
.551	-.327	.560	-1.032	.612
.585	-.280	.550	-1.090	.652
.592	-.260	.550	-1.449	-1.370
.613	-.127	.418	-1.288	-.900
.634	-.140	.272	-.795	-.612
.655	-.167	.133	-.449	-.323
.675	-.087	.020	-.276	-.112
.696	-.033	.040	-.179	-.033
.774	-.027	.199	[illegible]	.105
.852	-.127	.106	-.083	-.026
.930	-.033	.159	-.026	.132

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-8.298	-[illegible].820	-3.846	-3.551	-3.466
	.080	-2.002	-[illegible].581	-3.932	-3.564	-2.130
	.130	-3.260	-[illegible].898	-3.089	-2.763	-2.297
	.145	-9.602	-[illegible].977	-6.257	-6.115	-6.130
	.155	-4.070	-[illegible].224	-3.616	-3.231	-2.544
	.180	-2.654	-[illegible].885	-2.601	-2.276	-2.017
	.220	-1.633	-[illegible].056	-1.818	-1.641	-1.269
	.270	-1.363	-[illegible].611	-1.376	-1.224	-.928
	.400	-1.001	-[illegible].141	-.955	-.853	-.868
	.620	-1.001	-[illegible].041	-.790	-.385	-.908
	.685	-4.057	-[illegible].857	-.099	-1.327	-1.703
	.693	-4.063	-[illegible].042	-1.436	-1.526	-1.429
	.700	-2.700	-[illegible].611	-1.311	-1.154	-1.215
	.720	-1.436	.875	-.909	-.897	-.808
	.750	-1.008	.862	-.883	-.872	-.855
	.800	-.711	.862	-.856	-.801	-.835
	.900	-.454	.822	-.803	-.782	-.821
	.980	-.296	.769	-.777	-.750	-.788
Lower	.025	.790	.889	.856	.808	.714
	.120	.909	.875	.777	.763	.628
	.220	.869	.855	.830	.763	.701
	.300	.777	.802	.757	.731	.628
	.620	.790	.842	.790	.705	.341
	.750	.863	.889	.803	.705	.634
	.850	.692	.703	.593	.583	.514
	.950	.487	.365	.231	.250	.220

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TABLE 18 Continued (d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 6.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.3°											
.032	.288	.300	.277	.294	Upper	.010	.968	.918	.864	.857	.811
.053	.072	.075	.086	.073		.080	.522	.425	.392	.382	.403
.100	-.078	-.037	-.092	-.086		.130	-.287	-.512	-.453	-.382	-.391
.145	-.072	-.094	-.062	-.049		.145	-4.292	-3.984	-3.338	-3.600	-3.137
.189	.000	-.025	-.006	.000		.155	-1.566	-1.542	-1.280	-1.122	-.757
.234	-.024	.050	.000	.000		.180	-1.178	-.955	-.857	-.690	-.745
.280	-.030	.062	-.037	.031		.220	-.694	-.662	-.539	-.524	-.403
.326	-.024	.075	-.043	.061		.270	-.548	-.581	-.416	-.247	-.240
.371	-.078	.119	-.105	.135		.400	-.567	-.524	-.214	-.062	-.066
.392	-.036	.140	.031	.337		.620	-1.070	-.824	.576	.579	.709
.413	-.126	.150	-.092	-.269		.685	-6.062	-4.052	.147	-1.054	-1.605
.434	-.168	.137	-.308	-.386		.693	-5.979	-4.477	-1.096	-1.233	-1.322
.457	-.198	.150	-.407	-.245		.700	-3.738	-3.128	-.968	-.912	-1.142
.480	-.252	.180	-.370	-.092		.720	-1.764	-1.093	-.649	-.690	-.781
.502	-.337	.210	-.407	.073		.750	-1.184	-.637	-.643	-.740	-.799
.551	-.361	.240	-.579	.300		.800	-.783	-.606	-.643	-.746	-.805
.585	-.373	.268	-.678	.269		.900	-.535	-.556	-.612	-.746	-.787
.592	-.373	.268	-.783	-.778		.980	-.076	-.512	-.582	-.684	-.697
.613	-.282	.212	-.616	-.637	Lower	.025	-.427	-.231	-.037	.074	-.054
.634	-.240	.150	-.425	-.661		.120	-.439	-.187	-.067	.031	-.066
.655	-.222	.062	-.308	-.196		.220	-.388	-.225	-.086		-.072
.675	-.144	-.019	-.216	-.116		.300	-.140	-.287	-.141	-.055	-.060
.696	-.078	-.025	-.160	-.073		.620	.548	.337	-.067	-.160	-.162
.774	.024	.037	-.062	.043		.750	.834	.587	.006	-.074	-.012
.852	.006	-.056		-.110		.850	.700	.662	.263	.037	.084
.930	.072	-.187	.092	-.190		.950	.497	.375	.196		.012
α = 5.9°											
.032	.100	.500	.129	.362	Upper	.010	.700	.692	.674	.697	.681
.053	-.100	.250	-.071	.137		.080	-.070	-.077	-.069	-.065	.031
.100	-.194	.064	-.213	-.062		.130	-1.165	-1.295	-1.261	-1.187	-1.111
.145	-.137	-.006	-.168	-.037		.145	-6.266	-5.801	-5.045	-5.408	-4.727
.189	-.050	.051	-.110	.000		.155	-2.509	-2.442	-2.173	-2.013	-1.561
.234	-.087	.115	-.013	.000		.180	-1.853	-1.538	-1.461	-1.252	-1.286
.280	-.094	.115	.032	.000		.220	-1.127	-1.090	-.937	-.942	-.774
.326	-.094	.128	.019	.006		.270	-.860	-.897	-.699	-.549	-.500
.371	-.175	.186	-.194	.056		.400	-.732	-.718	-.350	-.226	-.237
.392	-.225	.230	-.213	.219		.620	-1.057	-.808	.524	.626	.699
.413	-.275	.276	-.503	.125		.685	-5.616	-3.891	.200	-1.058	-1.642
.434	-.325	.308	-.774	.169		.693	-5.451	-4.295	-1.130	-1.291	-1.336
.457	-.362	.325	-.736	.325		.700	-3.356	-3.051	-1.011	-.949	-1.186
.480	-.412	.340	-.652	.425		.720	-1.605	-1.058	-.662	-.716	-.774
.502	-.475	.375	-.620	.450		.750	-1.127	-.667	-.681	-.761	-.780
.551	-.487	.395	-.703	.487		.800	-.821	-.744	-.681	-.787	-.805
.585	-.468	.410	-.813	.531		.900	-.567	-.667	-.681	-.787	-.818
.592	-.443	.410	-.897	-.824		.980	-.127	-.686	-.649	-.749	-.755
.613	-.318	.333	-.684	-.637	Lower	.025	.019	.308	.331	.323	.175
.634	-.312	.231	-.471	-.437		.120	.096	.269	.300	.232	.094
.655	-.275	.115	-.342	-.112		.220	.478	.269	.262	.232	.106
.675	-.200	-.013	-.252	-.062		.300	.605	.397	.387	.284	.244
.696	-.137	-.045	-.187	-.050		.620	.694	.686	.649	.632	.393
.774	-.006	.051	-.065	.050		.750	.815	.788	.687	.684	.612
.852	-.019	.019	-.013	-.087		.850	.618	.590	.543	.536	.500
.930	.056	-.147	.065	-.137		.950	.452	.199	.187	.142	.181
α = 13.4°											
.032	-.089	.645	-.111	.259	Upper	.010	-.084	-.852	-.734	-.824	-.019
.053	-.236	.432	-.268	.063		.080	-.903	-.865	-.854	-.896	-.599
.100	-.197	.207	-.379	-.183		.130	-2.284	-2.381	-2.353	-2.211	-2.057
.145	-.146	.129	-.307	-.177		.145	-8.363	-7.886	-6.870	-7.339	-6.431
.189	-.089	.161	-.249	-.120		.155	-3.453	-3.556	-3.233	-3.068	-2.382
.234	-.089	.213	-.059	-.152		.180	-2.349	-2.246	-2.119	-1.930	-1.847
.280	-.121	.219	.072	-.158		.220	-1.471	-1.575	-1.417	-1.361	-1.121
.326	-.121	.232	.085	-.196		.270	-1.104	-1.233	-1.037	-.863	-.751
.371	-.248	.336	-.249	-.190		.400	-.903	-.884	-.519	-.353	-.369
.392	-.310	.390	-.563	-.139		.620	-.955	-.742	.392	.569	.675
.413	-.369	.439	-.883	.209		.685	-4.214	-3.478	.089	-1.053	-1.656
.434	-.414	.465	-1.190	.506		.693	-4.001	-3.820	-1.177	-1.276	-1.331
.457	-.427	.470	-.981	.595		.700	-2.562	-2.685	-1.050	-.922	-1.204
.480	-.439	.475	-.831	.557		.720	-1.413	-.916	-.721	-.720	-.815
.502	-.522	.478	-.759	.512		.750	-1.058	-.768	-.727	-.752	-.815
.551	-.458	.482	-.844	.531		.800	-.742	-.781	-.721	-.798	-.821
.585	-.433	.484	-.903	.614		.900	-.426	-.774	-.734	-.778	-.853
.592	-.408	.484	-.929	-.917		.980	-.194	-.774	-.709	-.739	-.802
.613	-.293	.374	-.713	-.531	Lower	.025	.452	.658	.645	.654	.503
.634	-.293	.265	-.491	-.557		.120	.800	.820	.753	.726	.586
.655	-.261	.181	-.327	-.114		.220	.774	.774	.734	.726	.637
.675	-.185	.019	-.222	-.089		.300	.652	.684	.645	.654	.535
.696	-.134	.000	-.150	-.051		.620	.723	.787	.727	.700	.344
.774	-.025	.123	-.013	.051		.750	.832	.871	.759	.720	.630
.852	-.057	.065	-.013	-.063		.850	.632	.632	.519	.549	.465
.930	.025	.000	.052	-.038		.950	.458	.232	.164	.157	.134

TABLE 18 Concluded
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
α = 19.0°				
.032	-.172	.763	-.283	.182
.053	-.298	.544	-.421	.013
.100	-.212	.318	-.487	-.247
.145	-.159	.245	-.461	-.273
.189	-.093	.252	-.402	-.208
.234	-.093	.305	-.092	-.266
.280	-.126	.298	.112	-.292
.326	-.172	.318	.059	-.370
.371	-.312	.418	-.342	-.429
.392	-.400	.475	-.810	-.500
.413	-.471	.537	-1.060	.195
.434	-.517	.570	-1.620	.617
.457	-.517	.560	-1.284	.689
.480	-.491	.550	-.994	.669
.502	-.531	.535	-.922	.637
.551	-.424	.525	-.922	.598
.585	-.371	.517	-1.047	.650
.592	-.345	.511	-1.317	-1.254
.613	-.225	.378	-1.060	-.800
.634	-.225	.285	-.626	-.656
.655	-.219	.146	-.342	-.234
.675	-.133	.013	-.211	-.058
.696	-.007	.033	-.138	-.013
.774	-.073	.199	-.007	.097
.852	-.013	.106	-.053	-.013
.930	-.007	.093	[illegible]	.091
α = 23.0°				
.032	-.276	.842	-.461	.047
.053	-.377	.650	-.546	-.108
.100	-.249	.424	-.617	-.363
.145	-.202	.305	-.552	-.410
.189	-.101	.338	-.520	-.343
.234	-.101	.378	-.136	-.410
.280	-.141	.365	.123	-.464
.326	-.222	.398	.058	-.551
.371	-.424	.484	-.435	-.612
.392	-.500	.545	-.929	-.800
.413	-.598	.603	-1.156	.195
.434	-.619	.630	-1.878	.666
.457	-.572	.610	-1.384	.740
.480	-.511	.590	-1.143	.706
.502	-.511	.570	-1.007	.659
.551	-.350	.550	-.968	.605
.585	-.329	.531	-1.033	.659
.592	-.303	.504	-1.351	-1.378
.613	-.182	.398	-1.410	-.753
.634	-.208	.259	-.929	-.699
.655	-.195	.119	-.526	-.303
.675	-.121	-.013	-.331	-.148
.696	-.087	.013	-.227	-.094
.774	-.061	.186	-.052	.067
.852	-.161	.086	-.084	-.061
.930	-.074	.146	-.039	.155

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
α = 19.0°						
Upper	.010	-3.460	-2.619	-2.859	-2.654	-2.507
	.080	-1.661	-1.618	-2.436	-2.147	-1.313
	.130	-3.029	-2.871	-2.807	-2.648	-2.871
	.145	-9.485	-8.654	-7.510	-7.785	-7.858
	.155	-4.062	-4.191	-3.872	-3.649	-3.269
	.180	-2.675	-2.745	-2.638	-2.358	-2.533
	.220	-1.668	-1.897	-1.800	-1.646	-1.671
	.270	-1.210	-1.485	-1.312	-1.120	-1.227
	.400	-.935	-1.008	-.650	-.474	-.716
	.620	-.975	-.829	-.013	.191	.550
	.685	-3.872	-2.241	.065	-1.080	-1.751
	.693	-3.774	-2.401	-1.260	-1.304	-1.399
	.700	-2.453	-1.731	-1.130	-.968	-1.220
	.720	-1.367	-.829	-.767	-.738	-.862
	.750	-1.020	-.902	-.780	-.771	-.869
	.800	-.759	-.855	-.786	-.863	-.882
	.900	-.491	-.836	-.767	-.757	-.908
	.980	-.301	-.802	-.747	-.724	-.836
Lower	.025	.726	.836	.819	.810	.683
	.120	.883	.855	.767	.757	.610
	.220	.837	.809	.799	.764	.643
	.300	.726	.743	.708	.698	.597
	.620	.778	.809	.754	.698	.318
	.750	.844	.862	.767	.738	.623
	.850	.654	.663	.546	.573	.491
	.950	.471	.279	.201	.224	.186
α = 23.0°						
Upper	.010	-7.725	-3.601	-3.745	-3.385	-3.295
	.080	-1.949	-3.309	-3.813	-3.391	-1.930
	.130	-3.196	-2.818	-2.965	-2.625	-2.246
	.145	-9.544	-7.984	-6.482	-6.334	-5.877
	.155	-4.035	-4.158	-3.644	-3.209	-2.387
	.180	-2.638	-2.805	-2.569	-2.222	-1.876
	.220	-1.611	-1.976	-1.768	-1.533	-1.143
	.270	-1.254	-1.519	-1.257	-1.104	-.814
	.400	-.936	-1.015	-.773	-.663	-.814
	.620	-.949	-.869	-.558	-.013	-.827
	.685	-3.950	-1.127	.020	-1.052	-1.594
	.693	-3.983	-1.373	-1.352	-1.306	-1.304
	.700	-2.631	-1.267	-1.244	-.962	-1.123
	.720	-1.410	-.889	-.847	-.747	-.733
	.750	-1.007	-.902	-.861	-.773	-.760
	.800	-.806	-.889	-.854	-.760	-.746
	.900	-.585	-.869	-.867	-.741	-.726
	.980	-.273	-.816	-.807	-.708	-.706
Lower	.025	.773	.889	.847	.838	.699
	.120	.897	.862	.787	.767	.612
	.220	.858	.855	.827	.786	.666
	.300	.780	.796	.746	.747	.619
	.620	.780	.849	.773	.708	.336
	.750	.858	.875	.793	.754	.612
	.850	.676	.690	.578	.604	.498
	.950	.494	.312	.202	.224	.222

L-927

L-927

TABLE 19 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 6.0; h_d/c = 3.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = −1.3°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.270	.292	.294	.312
.053	.060	.073	.098	.073
.100	-.090	-.037	-.098	-.061
.145	-.084	-.073	-.061	-.037
.189	.000	-.006	.012	.000
.234	-.030	.049	.012	.012
.280	-.042	.073	-.031	.037
.326	-.018	.091	-.024	.067
.371	-.078	.122	-.073	.135
.392	-.024	.135	.055	.331
.413	-.126	.146	-.086	-.245
.434	-.168	.146	-.312	-.343
.457	-.198	.160	-.416	-.220
.480	-.270	.185	-.374	-.049
.502	-.343	.110	-.398	.092
.551	-.373	.240	-.545	.300
.585	-.367	.268	-.661	.288
.592	-.355	.268	-.759	-.759
.613	-.270	.201	-.594	-.637
.634	-.234	.146	-.404	-.649
.655	-.204	.085	-.288	-.184
.675	-.138	-.006	-.190	-.098
.696	-.090	-.018	-.141	-.080
.774	.000	.055	-.061	.024
.852	-.006	-.043	.018	-.110
.930	.066	-.158	.086	-.220

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.974	.876	.864	.833	.793
	.080	.503	.402	.374	.386	.415
	.130	-.302	-.475	-.514	-.386	-.367
	.145	-4.242	-3.852	-3.448	-3.577	-3.119
	.155	-1.527	-1.479	-1.323	-1.102	-.757
	.180	-1.144	-.919	-.906	-.661	-.757
	.220	-.654	-.645	-.570	-.521	-.403
	.270	-.522	-.566	-.435	-.251	-.234
	.400	-.528	-.511	-.202	-.055	-.066
	.620	-1.024	-.791	.588	.600	.709
	.685	-5.920	-3.919	.171	-.992	-1.569
	.693	-5.813	-4.321	-1.072	-1.231	-1.304
	.700	-3.639	-3.031	-.968	-.882	-1.148
	.720	-1.722	-1.059	-.637	-.674	-.775
	.750	-1.144	-.621	-.655	-.735	-.751
	.800	-.760	-.584	-.649	-.747	-.763
	.900	-.484	-.554	-.606	-.723	-.751
	.980	-.075	-.499	-.594	-.619	-.673
Lower	.025	-.402	-.219	-.012	.061	-.054
	.120	-.408	-.189	-.031	.018	-.090
	.220	-.352	-.225	-.067	-.024	-.066
	.300	-.088	-.298	-.147	-.086	-.096
	.620	.578	.426	.092	-.049	-.150
	.750	.830	.663	.067	.024	.024
	.850	.704	.682	.343	.153	.114
	.950	.515	.347	.184	.098	.066

α = 5.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.075	.503	.171	.343
.053	-.119	.270	-.057	.119
.100	-.189	.082	-.183	-.081
.145	-.151	.019	-.152	-.050
.189	-.075	.057	-.082	-.012
.234	-.101	.119	-.013	-.012
.280	-.107	.126	.032	-.019
.326	-.101	.132	-.006	.006
.371	-.189	.195	-.152	.050
.392	-.230	.240	-.196	.219
.413	-.270	.289	-.481	.112
.434	-.314	.308	-.759	.150
.457	-.346	.320	-.696	.300
.480	-.402	.340	-.614	.418
.502	-.478	.360	-.588	.425
.551	-.478	.380	-.664	.468
.585	-.465	.396	-.772	.500
.592	-.440	.396	-.854	-.818
.613	-.327	.333	-.645	-.631
.634	-.321	.226	-.436	-.456
.655	-.295	.119	-.316	-.131
.675	-.201	-.013	-.247	-.087
.696	-.157	-.038	-.177	-.075
.774	-.013	.050	-.051	-.100
.852	-.038	.019	.000	-.106
.930	.038	-.138	.070	-.150

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.687	.679	.662	.709	.704
	.080	-.044	-.094	-.062	-.057	.031
	.130	-1.149	-1.326	-1.255	-1.113	-1.062
	.145	-6.150	-5.845	-4.945	-5.238	-4.613
	.155	-2.473	-2.470	-2.160	-1.955	-1.496
	.180	-1.798	-1.552	-1.436	-1.189	-1.244
	.220	-1.093	-1.087	-.918	-.873	-.735
	.270	-.812	-.892	-.687	-.493	-.452
	.400	-.699	-.710	-.325	-.164	-.214
	.620	-1.005	-.748	.556	.664	.710
	.685	-5.357	-3.746	.162	-1.025	-1.552
	.693	-5.189	-4.091	-1.111	-1.215	-1.320
	.700	-3.222	-2.960	-1.005	-.905	-1.156
	.720	-1.517	-1.012	-.674	-.683	-.817
	.750	-1.074	-.647	-.687	-.721	-.798
	.800	-.793	-.716	-.687	-.746	-.798
	.900	-.543	-.666	-.656	-.746	-.798
	.980	-.137	-.672	-.618	-.709	-.754
Lower	.025	.012	.314	.306	.310	.138
	.120	.112	.270	.281	.228	.088
	.220	.481	.283	.237	.202	.082
	.300	.606	.452	.343	.291	.239
	.620	.681	.691	.668	.721	.440
	.750	.812	.760	.755	.753	.679
	.850	.612	.591	.568	.588	.553
	.950	.450	.220	.181	.190	.195

α = 13.4°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.071	.639	-.085	.234
.053	-.247	.436	-.275	.032
.100	-.188	.221	-.373	-.195
.145	-.136	.127	-.314	-.195
.189	-.084	.164	-.255	-.169
.234	-.104	.209	-.052	-.175
.280	-.110	.228	.072	-.175
.326	-.117	.221	[illegible]	-.221
.371	-.253	.323	-.255	-.227
.392	-.300	.358	-.556	-.156
.413	-.357	.399	-.896	.188
.434	-.416	.462	-1.190	.507
.457	-.422	.363	-.981	.591
.480	-.435	.364	-.831	.552
.502	-.507	.366	-.752	.507
.551	-.455	.367	-.844	.533
.585	-.429	.468	-.890	.591
.592	-.416	.468	-.935	-.897
.613	-.286	.361	-.720	-.552
.634	-.292	.266	-.504	-.552
.655	-.253	.145	-.353	-.136
.675	-.175	.013	-.235	-.078
.696	-.130	.013	-.177	-.084
.774	-.019	.114	-.046	.006
.852	-.045	.063	-.039	-.091
.930	.032	-.013	.026	-.052

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.045	-.582	-.422	-.576	.019
	.080	-.852	-.797	-.832	-.877	-.624
	.130	-2.239	-2.265	-2.332	-2.159	-2.047
	.145	-8.299	-7.578	-6.874	-7.234	-6.477
	.155	-3.414	-3.378	-3.216	-3.029	-2.410
	.180	-2.336	-2.132	-2.124	-1.864	-1.839
	.220	-1.426	-1.480	-1.390	-1.289	-1.111
	.270	-1.065	-1.158	-1.014	-.811	-.721
	.400	-.871	-.841	-.474	-.294	-.331
	.620	-.858	-.595	.487	.634	.728
	.685	-4.156	-3.321	.143	-1.132	-1.592
	.693	-4.001	-3.650	-1.195	-1.289	-1.358
	.700	-2.568	-2.625	-1.065	-.968	-1.208
	.720	-1.387	-.886	-.715	-.733	-.845
	.750	-1.039	-.740	-.734	-.778	-.845
	.800	-.723	-.753	-.728	-.796	-.851
	.900	-.432	-.715	-.728	-.772	-.864
	.980	-.239	-.715	-.695	-.752	-.806
Lower	.025	.452	.658	.624	.648	.507
	.120	.787	.803	.741	.739	.624
	.220	.761	.746	.728	.720	.650
	.300	.652	.683	.637	.641	.546
	.620	.736	.772	.754	.746	.364
	.750	.839	.860	.780	.772	.689
	.850	.639	.626	.533	.602	.533
	.950	.458	.240	.182	.196	.162

TABLE 19 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.0$ $h_d/c = 3.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing , flap , or aileron				
α = 19.1°											
.032	-.174	.751	-.318	.133	Upper	.010	-3.690	-2.595	-2.725	-2.619	-2.149
.053	-.290	.553	-.451	-.053		.080	-1.618	-1.620	-2.162	-1.930	-1.039
.100	-.207	.329	-.511	-.272		.130	-2.989	-2.885	-2.845	-2.659	-2.736
.145	-.148	.237	-.464	-.279		.145	-9.349	-8.601	-7.546	-7.818	-7.466
.189	-.084	.244	-.411	-.239		.155	-3.957	-4.149	-3.853	-3.621	-3.039
.234	-.077	.290	-.106	-.305		.180	-2.631	-2.713	-2.606	-2.334	-2.336
.280	-.116	.310	.086	-.318		.220	-1.598	-1.877	-1.744	-1.618	-1.484
.326	-.142	.316	.060	-.385		.270	-1.182	-1.455	-1.267	-1.068	-1.052
.371	-.303	.415	-.332	-.431		.400	-.877	-.981	-.590	-.391	-.574
.392	-.375	.475	-.816	-.511		.620	-.884	- 744	.099	.305	.613
.413	-.445	.533	-1.061	.192		.685	-3.924	-1 989	.086	-1.054	-1.594
.434	-.484	.547	-1.585	.597		.693	-3.879	-2 134	-1.253	-1.300	-1.329
.457	-.484	.545	-1.293	.670		.700	-2.540	-1 620	-1.127	-.968	-1.155
.480	-.484	.540	-1.001	.643		.720	-1.351	- 883	-.769	-.736	-.813
.502	-.523	.530	-.908	.603		.750	-1.001	- 935	-.789	-.789	-.832
.551	-.413	.520	-.935	.570		.800	-.721	- 889	-.802	-.796	-.820
.585	-.355	.514	-1.015	.643		.900	-.494	-.863	-.789	-.776	-.832
.592	-.348	.501	-1.227	-1.187		.980	-.201	-.830	-.743	-.736	-.794
.613	-.226	.395	-1.048	-.800							
.634	-.226	.250	-.683	-.710	Lower	.025	.676	.830	.802	.796	.645
.655	-.219	.138	-.391	-.219		.120	.877	.869	.756	.729	.574
.675	-.129	.007	-.239	-.086		.220	.825	.823	.789	.749	.626
.696	-.103	[illegible]	-.153	-.053		.300	.734	.751	.703	.690	.574
.774	-.006	.158	-.033	.060		.620	.760	.817	.769	.736	.303
.852	-.077	.072	-.066	-.060		.750	.838	889	.829	.789	.665
.930	[illegible]	.079	-.007	.086		.850	.676	665	.550	.610	.516
						.950	.500	263	.225	.245	.200
α = 23.0°											
.032	-.235	.814	-.454	.059	Upper	.010	-7.719	-3 651	-3.642	-3.477	-3.322
.053	-.323	.598	-.566	-.112		.080	-1.950	-3 409	-3.721	-3.504	-1.856
.100	-.188	.410	-.626	-.362		.130	-3.225	-2 831	-2.878	-2.746	-2.481
.145	-.161	.303	-.593	-.395		.145	-9.575	-7 739	-6.369	-6.382	-6.536
.189	-.067	.323	-.520	-.323		.155	-4.040	-4 075	-3.576	-3.280	-2.723
.234	-.040	.356	-.132	-.402		.180	-2.671	-2 770	-2.522	-2.259	-2.138
.280	-.094	.370	.119	-.415		.220	-1.596	-1 930	-1.686	-1.561	-1.358
.326	-.188	.383	.020	-.533		.270	-1.289	-1 466	-1.205	-1.113	-1.002
.371	-.397	.504	-.421	-.599		.400	-.928	-.955	-.698	-.612	-1.056
.392	-.475	.550	-.968	-.757		.620	-.921	-.767	-.481	-.040	-.699
.413	-.558	.598	-1.212	.191		.685	-3.906	-.962	.020	-1.041	-1.513
.434	-.598	.625	-1.963	.678		.693	-3.926	-1.271	-1.324	-1.264	-1.278
.457	-.545	.610	-1.442	.738		.700	-2.618	-1 204	-1.185	-.922	-1.049
.480	-.457	.595	-1.185	.705		.720	-1.396	- 881	-.817	-.692	-.686
.502	-.457	.580	-1.041	.645		.750	-1.015	- 874	-.836	-.724	-.726
.551	-.296	.565	-.975	.606		.800	-.781	- 867	-.850	-.731	-.699
.585	-.249	.545	-1.034	.645		.900	-.581	- 841	-.803	-.711	-.679
.592	-.229	.531	-1.337	-1.258		.980	-.294	- 773	-.757	-.672	-.639
.613	-.128	.410	-1.297	-.790							
.634	-.148	.276	-.856	-.672	Lower	.025	.781	867	.850	.817	.713
.655	-.148	.114	-.514	-.283		.120	.908	854	.771	.757	.625
.675	-.081	.007	-.356	-.132		.220	.861	834	.823	.777	.672
.696	-.061	.013	-.224	-.059		.300	.788	780	.724	.711	.625
.774	-.020	.110	-.040	.072		.620	.808	841	.790	.764	.329
.852	-.114	.108	-.099	-.040		.750	.881	867	.836	.771	.679
.930	-.034	.155	-.053	.165		.850	.681	693	.586	.632	.572
						.950	.514	303	.231	.283	.282

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TABLE 19 Continued (b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 2.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	Fuselage, C_p values for spanwise stations, $\frac{y}{b/2}$, of: 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface	Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
α = -1.4°											
.032	.261	.296	.302	.323	Upper	.010	.962	.871	.840	.841	.806
.053	.050	.097	.073	.055		.080	.499	.399	.335	.351	.366
.100	-.087	-.042	-.073	-.079		.130	-.314	-.526	-.560	-.478	-.509
.145	-.099	-.079	-.048	-.030		.145	-4.296	-3.979	-3.578	-3.834	-3.548
.189	-.025	-.018	.012	.000		.155	-1.559	-1.554	-1.418	-1.270	-.993
.234	-.050	.060	.006	.018		.180	-1.159	-.974	-.986	-.792	-.931
.280	-.056	.073	-.024	.043		.220	-.672	-.695	-.663	-.647	-.565
.326	-.043	.085	-.024	.079		.270	-.542	-.623	-.529	-.381	-.385
.371	-.035	.115	.010	.140		.400	-.579	-.568	-.383	-.248	-.273
.392	-.025	.127	.042	.347		.620	-1.048	-.895	.170	.218	.254
.413	-.149	.139	-.103	-.158		.685	-5.960	-4.324	.164	-1.046	-1.749
.434	-.192	.163	-.321	-.310		.693	-5.893	-4.741	-1.065	-1.216	-1.470
.457	-.230	.185	-.423	-.213		.700	-3.704	-3.338	-.937	-.925	-1.303
.480	-.267	.205	-.369	-.030		.720	-1.769	-1.234	-.609	-.701	-.893
.502	-.366	.230	-.411	.103		.750	-1.183	-.732	-.651	-.720	-.875
.551	-.397	.250	-.562	.298		.800	-.783	-.575	-.627	-.762	-.893
.585	-.391	.272	-.689	.280		.900	-.499	-.605	-.584	-.732	-.819
.592	-.378	.272	-.810	-.822		.980	-.037	-.514	-.578	-.683	-.695
.613	-.304	.194	-.647	-.639	Lower	.025	-.345	-.169		.097	-.050
.634	-.248	.157	-.441	-.688		.120	-.351	-.145	-.006	.030	-.093
.655	-.211	.097	-.308	-.176		.220	-.296	-.163	-.037	.018	-.056
.675	-.124	.006	-.212	-.091		.300	-.043	-.224	-.103	-.042	-.087
.696	-.074	.012	-.151	-.043		.620	.555	.375	.097	-.024	-.174
.774	-.012	.054	-.048	.049		.750	.814	.599	.103	.054	
.852	-.012	-.060	-.006	-.110		.850	.709	.653	.329	.157	.130
.930	.074	-.151	.103	-.213		.950	.536	.375	.189	.115	.099
α = 5.8°											
.032	.071	.506	.152	.327	Upper	.010	.666	.660	.616	.652	.673
.053	-.103	.314	-.070	.119		.080	-.088	-.122	-.151	-.152	-.051
.100	-.179	.083	-.171	-.094		.130	-1.188	-1.397	-1.414	-1.290	-1.250
.145	-.141	.013	-.145	-.069		.145	-6.278	-6.045	-5.380	-5.744	-5.154
.189	-.051	.058	-.089	-.006		.155	-2.545	-2.603	-2.420	-2.220	-1.782
.234	-.090	.115	-.006	-.031		.180	-1.823	-1.660	-1.634	-1.411	-1.462
.280	-.090	.122	.057	-.025		.220	-1.125	-1.160	-1.094	-1.075	-.923
.326	-.090	.135	-.013	-.019		.270	-.848	-.981	-.861	-.702	-.667
.371	-.192	.199	-.177	.019		.400	-.723	-.801	-.572	-.430	-.436
.392	-.240	.240	-.221	.201		.620	-1.043	-.917	.151	.202	.244
.413	-.282	.276	-.493	.132		.685	-5.556	-4.429	.119	-1.132	-1.808
.434	-.346	.314	-.772	.163		.693	-5.405	-4.865	-1.200	-1.322	-1.506
.457	-.372	.330	-.715	.308		.700	-3.343	-3.468	-1.068	-1.012	-1.327
.480	-.429	.350	-.652	.421		.720	-1.596	-1.295	-.735	-.765	-.923
.502	-.506	.370	-.620	.434		.750	-1.156	-.821	-.735	-.791	-.885
.551	-.487	.390	-.721	.471		.800	-.848	-.769	-.779	-.848	-.859
.585	-.462	.404	-.848	.515		.900	-.553	-.737	-.685	-.778	-.872
.592	-.436	.397	-.930	-.899		.980	-.082	-.699	-.660	-.746	-.814
.613	-.321	.327	-.715	-.654	Lower	.025	-.107	.327	.327	.348	.154
.634	-.295	.231	-.493	-.522		.120	.207	.282	.283	.240	.077
.655	-.256	.135	-.329	-.119		.220	.591	.276	.277	.240	.077
.675	-.167	.013	-.247	-.050		.300	.597	.436	.452	.386	.327
.696	-.109	-.006	-.177	-.044		.620	.698	.705	.672	.683	.417
.774	-.013	.064	-.044	.050		.750	.811	.782	.723	.702	.673
.852	-.013	.032	.	-.113		.850	.622	.609	.559	.550	.558
.930	.077	-.122	.070	-.170		.950	.478	.250	.176	.171	.160
α = 13.4°											
.032	-.090	.637	-.076	.263	Upper	.010	-.121	-1.034	-1.135	-1.006	-.084
.053	-.258	.411	-.248	.045		.080	-.975	-.922	-.955	-.949	-.703
.100	-.194	.192	-.299	-.167		.130	-2.448	-2.546	-2.532	-2.318	-2.213
.145	-.155	.113	-.293	-.147		.145	-8.822	-8.289	-7.256	-7.571	-6.853
.189	-.090	.146	-.255	-.128		.155	-3.692	-3.740	-3.487	-3.260	-2.633
.234	-.097	.206	-.064	-.154		.180	-2.515	-2.394	-2.321	-2.057	-2.046
.280	-.116	.212	.083	-.160		.220	-1.567	-1.678	-1.571	-1.477	-1.278
.326	-.123	.239	.032	-.218		.270	-1.190	-1.326	-1.186	-1.012	-.903
.371	-.271	.345	-.261	-.212		.400	-.995	-.995	-.724	-.579	-.574
.392	-.310	.385	-.548	-.154		.620	-.982	-.935	.083	.127	.207
.413	-.368	.424	-.885	.179		.685	-3.254	-4.145	.051	-1.114	-1.762
.434	-.445	.464	-1.197	.500		.693	-2.932	-4.529	-1.224	-1.318	-1.458
.457	-.432	.465	-.981	.590		.700	-1.856	-3.223	-1.128	-1.000	-1.284
.480	-.458	.467	-.841	.551		.720	-1.022	-1.167	-.782	-.745	-.910
.502	-.510	.465	-.758	.513		.750	-.901	-.763	-.769	-.796	-.891
.551	-.452	.464	-.802	.551		.800	-.767	-.836	-.769	-.860	-.884
.585	-.426	.464	-.834	.596		.900	-.585	-.836	-.763	-.777	-.891
.592	-.400	.464	-.821	-.801		.980	-.491	-.809	-.731	-.745	-.839
.613	-.284	.358	-.713	-.600	Lower	.025	.444	.690	.654	.650	.503
.634	-.277	.265	-.509	-.564		.120	.820	.836	.744	.713	.607
.655	-.258	.133	-.344	-.378		.220	.793	.769	.737	.707	.626
.675	-.168	.	-.210	-.154		.300	.659	.696	.667	.618	.516
.696	-.142	.013	-.134	-.051		.620	.746	.796	.731	.707	.329
.774	-.006	.133	-.075	.064		.750	.841	.875	.763	.732	.658
.852	-.045	.080	-.013	-.064		.850	.612	.650	.538	.554	.484
.930	.019	-.020	.045	-.051		.950	.383	.279	.154	.185	.129

TABLE 19 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 2.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = 19.0°$

x/l	Fuselage 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.170	.729	-.325	.146
.053	-.307	.544	-.455	-.040
.100	-.216	.325	-.526	-.298
.145	-.170	.225	-.468	-.298
.189	-.098	.245	-.403	-.252
.234	-.085	.292	-.117	-.285
.280	-.131	.312	.110	-.312
.326	-.157	.318	.078	-.405
.371	-.340	.418	-.286	-.444
.392	-.400	.470	-.799	-.524
.413	-.464	.517	-1.065	.172
.434	-.543	.517	-1.618	.623
.457	-.510	.517	-1.293	.683
.480	-.504	.517	-1.014	.630
.502	-.523	.517	-.936	.623
.551	-.432	.517	-.923	.590
.585	-.399	.517	-.988	.670
.592	-.366	.504	-1.195	-1.214
.613	-.242	.385	-1.052	-.700
.634	-.229	.252	-.676	-.544
.655	-.209	.113	-.377	-.444
.675	-.111	-.013	-.195	-.206
.696	-.085	.013	-.104	-.020
.774	-.082	.050	.039	.099
.852	-.078	.093	-.045	-.020
.930	-.007	.080	.	.093

Surface	x/c	Wing, flap, or aileron 0.221	0.425	0.640	0.800	0.918
Upper	.010	-3.931	-2.745	-2.977	-2.833	-2.590
	.080	-1.683	-1.777	-2.546	-2.293	-1.347
	.130	-3.073	-2.984	-2.951	-2.807	-3.022
	.145	-9.531	-8.846	-7.825	-8.154	-8.163
	.155	-4.074	-4.310	-4.098	-3.898	-3.473
	.180	-2.709	-2.871	-2.818	-2.579	-2.708
	.220	-1.676	-2.009	-1.943	-1.826	-1.805
	.270	-1.228	-1.565	-1.466	-1.306	-1.400
	.400	-.942	-1.127	-.889	-.728	-.975
	.620	-.910	-1.068	-.232	-.110	.052
	.685	-3.164	-3.488	-.086	-1.104	-1.740
	.693	-3.008	-3.727	-1.439	-1.390	-1.478
	.700	-1.923	-2.659	-1.306	-1.027	-1.269
	.720	-.968	-1.04	-.915	-.812	-.948
	.750	-.747	-.81	-.955	-.897	-.903
	.800	-.630	-.91	-.948	-.884	-.916
	.900	-.513	-.94	-.836	-.754	-.890
	.980	-.429	-.88	-.855	-.715	-.837
Lower	.025	.669	.83	.802	.799	.667
	.120	.884	.85	.776	.741	.576
	.220	.838	.822	.789	.741	.634
	.300	.741	.743	.710	.689	.589
	.620	.767	.809	.776	.728	.314
	.750	.838	.849	.816	.760	.628
	.850	.656	.65	.564	.572	.510
	.950	.422	.31	.199	.247	.196

$\alpha = 23.1°$

x/l	Fuselage 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.235	.813	-.467	.033
.053	-.353	.609	-.569	-.114
.100	-.222	.433	-.616	-.354
.145	-.183	.298	-.596	-.401
.189	-.098	.325	-.528	-.347
.234	-.065	.345	-.142	-.407
.280	-.118	.379	.129	-.441
.326	-.209	.379	.102	-.548
.371	-.406	.474	-.440	-.614
.392	-.475	.530	-.982	-.781
.413	-.589	.596	-1.273	.214
.434	-.615	.623	-2.038	.654
.457	-.549	.600	-1.510	.741
.480	-.491	.580	-1.239	.708
.502	-.484	.560	-1.097	.668
.551	-.334	.540	-1.050	.614
.585	-.275	.528	-1.124	.668
.592	-.255	.521	-1.476	-1.376
.613	-.164	.393	-1.395	-.850
.634	-.177	.257	-.894	-.674
.655	-.177	.115	-.501	-.300
.675	-.098	.007	-.311	-.120
.696	-.059	.027	-.203	-.053
.774	-.039	.217	-.027	.080
.852	-.118	.095	-.081	-.027
.930	-.052	.135	-.041	.147

Surface	x/c	Wing, flap, or aileron 0.221	0.425	0.640	0.800	0.918
Upper	.010	-7.798	-3.79	-3.833	-3.663	-3.297
	.080	-2.002	-3.54	-3.913	-3.717	-1.871
	.130	-3.260	-2.93	-3.072	-2.912	-2.263
	.145	-9.668	-7.93	-6.370	-6.257	-5.985
	.155	-4.083	-4.22	-3.653	-3.318	-2.427
	.180	-2.687	-2.89	-2.604	-2.323	-1.936
	.220	-1.627	-2.05	-1.803	-1.639	-1.184
	.270	-1.324	-1.59	-1.329	-1.232	-.877
	.400	-.975	-1.11	-.901	-.860	-.850
	.620	-.988	-.96	-.721	-.379	-.922
	.685	-4.103	-1.52	-.013	-1.111	-1.563
	.693	-4.116	-1.69	-1.376	-1.408	-1.315
	.700	-2.727	-1.44	-1.242	-1.022	-1.112
	.720	-1.462	-.88	-.901	-.792	-.733
	.750	-1.047	-.90	-.881	-.867	-.746
	.800	-.777	-.89	-.875	-.813	-.700
	.900	-.520	-.85	-.828	-.779	-.720
	.980	-.310	-.81	-.788	-.731	-.680
Lower	.025	.784	.87	.848	.840	.687
	.120	.896	.85	.775	.799	.615
	.220	.856	.82	.828	.806	.667
	.300	.784	.77	.735	.752	.589
	.620	.784	.84	.795	.772	.314
	.750	.869	.86	.835	.819	.621
	.850	.672	.67	.601	.670	.510
	.950	.481	.32	.247	.311	.249

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TABLE 19 Continued
(c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 1.0 h_d/c = 0.5

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.7°											
.032	.271	.308	.258	.269	Upper	.010	.906	.861	.821	.823	.813
.053	.045	.082	.057	.067		.080	.398	.295	.239	.226	.265
.100	-.103	-.025	-.094	-.080		.130	-.447	-.716	-.827	-.779	-.787
.145	-.097	-.069	-.069	-.043		.145	-4.544	-4.550	-4.305	-4.827	-4.420
.189	-.032	-.019	-.038	-.018		.155	-1.696	-1.879	-1.868	-1.804	-1.426
.234	-.071	.044	.006	.000		.180	-1.280	-1.238	-1.354	-1.232	-1.323
.280	-.077	.082	-.013	.024		.220	-.784	-.943	-.986	-1.043	-.897
.326	-.058	.094	-.038	.073		.270	-.637	-.861	-.888	-.786	-.729
.371	-.116	.126	-.101	.050		.400	-.686	-.886	-.937	-.867	-.729
.392	-.032	.140	[illegible]	.025		.620	-1.231	-1.515	-1.292	-1.357	-1.045
.413	-.200	.157	-.163	[illegible]		.685	-6.572	-6.718	-3.558	-4.933	-4.440
.434	-.239	.195	-.415	-.135		.693	-6.486	-7.246	-5.010	-5.178	-3.698
.457	-.284	.205	-.540	-.067		.700	-4.159	-5.248	-3.822	-3.475	-2.994
.480	-.342	.215	-.515	-.472		.720	-2.021	-2.306	-1.715	-1.760	-1.387
.502	-.432	.225	-.547	-.440		.750	-1.366	-1.414	-1.053	-1.131	-1.097
.551	-.471	.233	-.465	-.435		.800	-.937	-.811	-.717	-.710	-.936
.585	-.458	.245	-.892	-.851		.900	-.582	-.245	-.410	-.427	-.807
.592	-.426	.263	-1.050	-.931		.980	-.006	.207	-.196	-.258	-.665
.613	-.316	.264	-.302	-.717	Lower	.025	-.116	-.044	.122	.157	-.019
.634	-.271	.195	-.270	-.717		.120	-.184	.000	.104	.082	-.019
.655	-.213	.132	-.434	-.178		.220	-.116	-.031	.061	.057	-.032
.675	-.129	.038	-.277	-.055		.300	.043	-.069	-.006	.013	-.058
.696	-.065	-.031	-.195	-.031		.620	.533	.427	.227	.119	-.168
.774	-.058	.006	-.075	.006		.750	.766	.622	.263	.214	.136
.852	-.019	-.019	-.006	-.129		.850	.723	.723	.441	.339	.252
.930	.071	-.207	.088	-.263		.950	.557	.597	.361	.283	.200
α = 5.5°											
.032	.072	.471	.114	.336	Upper	.010	.561	.516	.441	.449	.520
.053	-.105	.239	-.095	.092		.080	-.245	-.297	-.428	-.455	-.329
.100	-.204	.065	-.221	-.105		.130	-1.452	-1.678	-1.943	-1.841	-1.844
.145	-.145	-.013	-.171	-.171		.145	-7.034	-6.718	-6.691	-7.224	-6.751
.189	-.072	.052	-.070	-.059		.155	-2.891	-2.975	-3.168	-3.068	-2.628
.234	-.112	.116	-.019	-.046		.180	-2.142	-1.949	-2.252	-2.069	-2.160
.280	-.112	.129	.032	-.033		.220	-1.355	-1.439	-1.640	-1.695	-1.515
.326	-.112	.142	.013	-.033		.270	-1.033	-1.220	-1.403	-1.290	-1.225
.371	-.217	.226	-.196	.007		.400	-.916	-1.104	-1.324	-1.259	-1.199
.392	-.260	.270	-.304	.171		.620	-1.162	-1.626	-1.554	-1.759	-1.660
.413	-.303	.303	-.582	.244		.685	-3.878	-6.615	-3.747	-5.858	-8.483
.434	-.375	.348	-.905	.277		.693	-3.446	-7.015	-5.209	-6.408	-8.035
.457	-.421	.345	-.860	.389		.700	-2.026	-5.117	-3.952	-4.397	-6.415
.480	-.468	.340	-.759	-.310		.720	-.929	-2.246	-1.752	-2.353	-3.359
.502	-.527	.335	-.734	-.500		.750	-.787	-1.329	-1.133	-1.480	-2.450
.551	-.514	.330	-.829	-.823		.800	-.703	-.768	-.889	-.993	-1.837
.585	-.494	.328	-.886	-.876		.900	-.581	-.265	-.757	-.734	-1.093
.592	-.468	.325	-.854	-.889		.980	-.458	.123	-.547	-.468	-.553
.613	-.310	.323	-.690	-.600	Lower	.025	.174	.407	.356	.411	.191
.634	-.316	.226	-.538	-.560		.120	.232	.361	.316	.272	.066
.655	-.263	.123	-.392	-.487		.220	.523	.445	.553	.418	.329
.675	-.158	.013	-.266	-.191		.300	.594	.568	.626	.563	.487
.696	-.112	-.045	-.177	-.026		.620	.710	.710	.698	.645	.132
.774	-.020	[illegible]	-.013	.033		.750	.807	.774	.724	.677	.593
.852	-.046	.052	-.038	-.125		.850	.607	.658	.606	.557	.487
.930	.033	-.110	.019	-.125		.950	.368	.490	.303	.285	.237
α = 13.3°											
.032	-.072	.621	-.088	.250	Upper	.010	-.307	-1.537	-1.756	-1.804	-1.498
.053	-.229	.392	-.270	.051		.080	-1.112	-1.112	-1.186	-1.276	-1.158
.100	-.209	.190	-.358	-.167		.130	-2.597	-2.852	-3.064	-2.985	-3.081
.145	-.157	.098	-.308	-.263		.145	-8.987	-8.896	-8.417	-9.094	-8.765
.189	-.098	.137	-.245	-.263		.155	-3.800	-4.127	-4.218	-4.230	-3.696
.234	-.111	.183	-.044	-.167		.180	-2.610	-2.715	-2.917	-2.797	-2.885
.280	-.137	.203	.075	-.192		.220	-1.701	-1.943	-2.096	-2.162	-1.995
.326	-.144	.229	.044	-.231		.270	-1.295	-1.596	-1.731	-1.665	-1.616
.371	-.281	.327	-.245	-.237		.400	-1.079	-1.282	-1.436	-1.433	-1.498
.392	-.240	.375	-.610	-.205		.620	-1.184	-1.688	-1.615	-1.816	-1.930
.413	-.392	.425	-.949	.218		.685	-2.780	-5.900	-3.423	-5.863	-9.563
.434	-.464	.471	-1.251	.545		.693	-2.270	-6.240	-4.731	-6.027	-9.007
.457	-.477	.460	-1.087	.609		.700	-1.406	-4.454	-3.551	-4.160	-7.156
.480	-.477	.455	-.936	.571		.720	-.674	-1.871	-1.577	-2.181	-3.813
.502	-.582	.440	-.880	.526		.750	-.602	-1.060	-1.000	-1.370	-2.747
.551	-.504	.425	-.911	-.936		.800	-.491	-.563	-.744	-1.056	-2.060
.585	-.464	.410	-.886	-.904		.900	-.412	-.203	-.827	-.999	-1.400
.592	-.425	.395	-.874	-.885		.980	-.425	-.111	-.718	-.924	-1.040
.613	-.355	.373	-.672	-.128	Lower	.025	.491	.706	.712	.698	.556
.634	-.294	.268	-.490	-.359		.120	.831	.831	.744	.698	.543
.655	-.249	.124	-.339	-.436		.220	.785	.765	.744	.698	.576
.675	-.164	-.007	-.233	-.282		.300	.680	.713	.673	.616	.471
.696	-.098	-.111	-.119	-.109		.620	.765	.791	.737	.660	.039
.774	-.007	.059	.019	.064		.750	.844	.896	.737	.698	.523
.852	-.072	.078	-.057	-.071		.850	.608	.713	.603	.534	.406
.930	.013	.026	.006	-.006		.950	.386	.504	.295	.245	.144

TABLE 19 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.5$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					$\alpha = 18.7°$						
.032	-.196	.733	-.290	.140	Upper	.010	-5.017	-3.281	-3.819	-3.695	-3.565
.053	-.327	.504	-.435	-.040		.080	-1.845	-2.380	-3.666	-3.570	-2.178
.100	-.209	.303	-.527	-.300		.130	-3.231	-3.140	-3.359	-3.372	-3.532
.145	-.183	.195	-.474	-.327		.145	-9.877	-9.266	-8.640	-8.996	-9.465
.189	-.098	.235	-.415	-.294		.155	-4.265	-4.666	-4.781	-4.702	-4.317
.234	-.098	.276	-.092	-.341		.180	-2.878	-3.147	-3.459	-3.300	-3.414
.280	-.144	.303	.092	-.374		.220	-1.779	-2.293	-2.524	-2.503	-2.473
.326	-.183	.323	.020	-.467		.270	-1.387	-1.822	-2.070	-1.982	-2.034
.371	-.353	.410	-.310	-.507		.400	-1.040	-1.392	-1.629	-1.614	-1.845
.392	-.430	.470	-.823	-.648		.620	-1.007	-1.634	-1.709	-1.805	-1.818
.413	-.517	.545	-1.093	.147		.685	-2.754	-4.714	-2.878	-4.847	-7.725
.434	-.569	.572	-1.739	.628		.693	-2.603	-5.110	-4.267	-5.005	-6.947
.457	-.556	.560	-1.383	.701		.700	-1.596	-3.651	-3.239	-3.576	-5.423
.480	-.523	.570	-1.120	.654		.720	-.759	-1.526	-1.509	-1.910	-2.976
.502	-.549	.580	-1.027	.641		.750	-.602	-.854	-1.028	-1.317	-2.185
.551	-.432	.598	-.988	.608		.800	-.530	-.430	-.761	-1.047	-1.681
.585	-.406	.550	-1.054	-1.115		.900	-.406	-.222	-.821	-.869	-1.256
.592	-.360	.480	-1.185	-1.189		.980	-.340	-.087	-.775	-.771	-.916
.613	-.222	.410	-.896	-.520							
.634	-.229	.289	-.593	-.387	Lower	.025	.680	.841	.815	.803	.615
.655	-.203	.155	-.356	-.387		.120	.863	.847	.748	.718	.504
.675	-.111	.054	-.211	-.240		.220	.811	.820	.801	.771	.582
.696	-.065	-.054	-.079	-.080		.300	.733	.760	.708	.705	.523
.774	-.013	.121	.066	-.055		.620	.765	.841	.768	.685	.085
.852	-.098	.101	-.053	-.033		.750	.831	.888	.714	.692	.536
.930	-.033	.114	-.020	.093		.850	.654	.746	.626	.580	.412
						.950	.438	.518	.294	.270	.170
					$\alpha = 22.9°$						
.032	-.254	.803	-.445	.039	Upper	.010	-8.959	-4.044	-4.284	-4.014	-3.906
.053	-.347	.612	-.555	-.124		.080	-2.174	-3.813	-4.422	-4.033	-2.638
.100	-.227	.395	-.594	-.373		.130	-3.494	-3.016	-3.408	-3.459	-2.644
.145	-.194	.290	-.561	-.406		.145	-10.265	-8.235	-7.123	-6.531	-6.931
.189	-.093	.316	-.497	-.347		.155	-4.395	-4.426	-4.219	-3.756	-3.058
.234	-.080	.356	-.123	-.425		.180	-2.912	-3.076	-3.133	-2.775	-2.444
.280	-.140	.362	.123	-.451		.220	-1.781	-2.246	-2.296	-2.078	-1.603
.326	-.227	.382	.097	-.569		.270	-1.476	-1.791	-1.851	-1.671	-1.242
.371	-.441	.487	-.432	-.661		.400	-1.111	-1.350	-1.419	-1.342	-1.189
.392	-.530	.530	-1.000	-.857		.620	-1.029	-1.429	-1.315	-1.142	-1.122
.413	-.614	.599	-1.271	.177		.685	-3.866	-3.688	-.831	-1.962	-2.090
.434	-.654	.632	-1.988	.661		.693	-3.806	-4.070	-2.152	-2.123	-1.729
.457	-.594	.634	-1.549	.726		.700	-2.533	-2.931	-1.812	-1.510	-1.529
.480	-.528	.636	-1.304	.700		.720	-1.259	-1.284	-1.132	-1.162	-1.095
.502	-.521	.638	-1.149	.661		.750	-.853	-.797	-1.040	-1.110	-1.095
.551	-.347	.639	-1.065	.608		.800	-.616	-.527	-.968	-1.026	-1.088
.585	-.287	.639	-1.174	.641		.900	-.481	-.349	-.870	-.968	-1.022
.592	-.267	.510	-1.471	-1.478		.980	-.433	-.165	-.739	-.974	-.962
.613	-.147	.402	-1.155	-.700							
.634	-.160	.290	-.716	-.497	Lower	.025	.833	.869	.844	.826	.681
.655	-.154	.145	-.413	-.379		.120	.934	.869	.772	.774	.608
.675	-.080	.033	-.213	-.157		.220	.894	.843	.811	.794	.674
.696	-.047	.007	-.110	-.020		.300	.819	.803	.765	.742	.594
.774	-.027	.125	.039	.111		.620	.833	.836	.765	.716	.267
.852	-.114	.112	-.065	.013		.750	.907	.883	.785	.742	.588
.930	-.053	.125	-.032	.124		.850	.711	.738	.621	.587	.481
						.950	.501	.494	.268	.239	.160

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TABLE 19 Continued (d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = -1.5°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.247	.314	.304	.302
.053	.049	.079	.061	.062
.100	-.111	-.036	-.097	-.080
.145	-.086	-.085	-.043	-.037
.189	-.031	-.018	.012	-.006
.234	-.055	.048	.018	.018
.280	-.055	.073	-.018	.043
.326	-.049	.085	-.012	.074
.371	-.111	.121	-.097	.136
.392	-.043	.132	.012	.357
.413	-.160	.145	-.134	-.123
.434	-.216	.163	-.365	-.247
.457	-.234	.180	-.481	-.179
.480	-.302	.200	-.438	-.012
.502	-.394	.220	-.463	.123
.551	-.413	.230	-.651	.308
.585	-.407	.236	-.779	.259
.592	-.388	.225	-.913	-.962
.613	-.296	.218	-.749	-.696
.634	-.259	.163	-.523	-.740
.655	-.210	.097	-.353	-.173
.675	-.129	.024	-.225	-.080
.696	-.074	-.060	-.158	-.031
.774	-.049	.012	-.061	.025
.852	-.025	-.024	.024	-.123
.930	.062	-.181	.110	-.228

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.949	.883	.838	.858	.814
	.080	.475	.375	.314	.316	.357
	.130	-.353	-.568	-.666	-.627	-.567
	.145	-4.357	-4.155	-3.908	-4.296	-3.785
	.155	-1.594	-1.615	-1.603	-1.491	-1.134
	.180	-1.193	-1.052	-1.134	-.974	-1.054
	.220	-.706	-.762	-.795	-.803	-.672
	.270	-.572	-.677	-.690	-.560	-.487
	.400	-.596	-.653	-.592	-.487	-.425
	.620	-1.108	-1.052	-.271	-.231	-.247
	.685	-6.147	-4.953	.068	-1.290	-1.960
	.693	-6.892	-5.455	-1.282	-1.436	-1.615
	.700	-3.846	-3.858	-1.153	-1.120	-1.448
	.720	-1.832	-1.506	-.820	-.876	-1.072
	.750	-1.229	-.865	-.764	-.907	-1.023
	.800	-.815	-.593	-.727	-.791	-.881
	.900	-.517	-.556	-.567	-.694	-.721
	.980	-.061	-.472	-.555	-.639	-.653
Lower	.025	-.256	-.133	.031	.079	-.049
	.120	-.304	-.103	.025	.024	-.086
	.220	-.262	-.139	-.006	.000	-.068
	.300	-.024	-.194	-.080	-.049	-.080
	.620	.523	.387	.049	.012	-.166
	.750	.791	.605	.154	.122	.049
	.850	.712	.647	.345	.262	.160
	.950	.548	.423	.222	.164	.123

$\alpha = 5.7°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.088	.490	.126	.310
.053	-.101	.255	-.082	.095
.100	-.176	.070	-.226	-.120
.145	-.145	.013	-.170	-.082
.189	-.063	.064	-.094	-.032
.234	-.094	.140	.006	-.038
.280	-.088	.134	.050	-.044
.326	-.088	.153	.031	-.025
.371	-.182	.217	-.195	.006
.392	-.230	.260	-.245	.040
.413	-.289	.293	-.534	.082
.434	-.333	.331	-.823	.253
.457	-.365	.350	-.786	.386
.480	-.434	.370	-.698	.436
.502	-.509	.390	-.679	.436
.551	-.496	.410	-.798	.462
.585	-.465	.439	-.930	.512
.592	-.440	.400	-1.006	-.981
.613	-.302	.357	-.773	-.601
.634	-.289	.255	-.515	-.601
.655	-.251	.153	-.339	-.108
.675	-.151	.038	-.220	-.025
.696	-.101	-.038	-.145	-.006
.774	-.025	.013	-.038	.044
.852	-.025	.045	.013	-.101
.930	.050	-.121	.075	-.158

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.628	.605	.576	.610	.641
	.080	-.160	-.223	-.196	-.226	-.113
	.130	-1.321	-1.535	-1.550	-1.427	-1.395
	.145	-6.692	-6.381	-5.630	-6.146	-5.593
	.155	-2.744	-2.757	-2.562	-2.470	-1.986
	.180	-1.994	-1.796	-1.784	-1.621	-1.647
	.220	-1.237	-1.286	-1.265	-1.276	-1.094
	.270	-.955	-1.095	-1.031	-.918	-.836
	.400	-.853	-.949	-.848	-.735	-.666
	.620	-1.237	-1.210	-.455	-.408	-.365
	.685	-5.756	-5.432	-.323	-1.452	-2.357
	.693	-5.538	-5.922	-1.619	-1.665	-1.948
	.700	-3.455	-4.215	-1.430	-1.269	-1.766
	.720	-1.705	-1.713	-1.018	-1.012	-1.225
	.750	-1.263	-1.025	-.911	-.987	-1.087
	.800	-.949	-.751	-.765	-.855	-.943
	.900	-.500	-.751	-.652	-.779	-.880
	.980	-.147	-.675	-.639	-.798	-.880
Lower	.025	-.026	.395	.342	.346	.170
	.120	.295	.318	.285	.195	.088
	.220	.628	.395	.335	.239	.138
	.300	.603	.522	.500	.440	.434
	.620	.705	.720	.639	.660	.377
	.750	.827	.802	.702	.691	.654
	.850	.615	.637	.500	.503	.515
	.950	.481	.325	.158	.145	.138

$\alpha = 13.2°$

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.077	.641	-.078	.268
.053	-.244	.414	-.260	.046
.100	-.186	.207	-.364	-.177
.145	-.141	.107	-.305	-.177
.189	-.090	.154	-.253	-.124
.234	-.096	.200	-.045	-.150
.280	-.128	.220	.084	-.164
.326	-.154	.227	.019	-.216
.371	-.276	.314	-.266	-.216
.392	-.325	.365	-.572	-.177
.413	-.372	.427	-.929	.203
.434	-.449	.481	-1.254	.523
.457	-.442	.475	-1.046	.589
.480	-.462	.472	-.916	.556
.502	-.538	.469	-.838	.530
.551	-.487	.468	-.877	.563
.585	-.436	.467	-.884	.608
.592	-.391	.420	-.884	-.824
.613	-.288	.381	-.715	-.500
.634	-.282	.240	-.533	-.432
.655	-.250	.134	-.344	-.438
.675	-.147	.013	-.195	-.235
.696	-.115	-.087	-.091	-.052
.774	-.006	.060	.052	.092
.852	-.051	.060	-.019	-.059
.930	.019		.032	-.007

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.182	-1.255	-1.465	-1.390	-.667
	.080	-1.001	-1.015	-1.125	-1.169	-.897
	.130	-2.449	-2.704	-2.872	-2.716	-2.596
	.145	-8.693	-8.707	-8.006	-8.582	-7.660
	.155	-3.645	-3.986	-3.931	-3.859	-3.038
	.180	-2.514	-2.564	-2.675	-2.482	-2.372
	.220	-1.579	-1.823	-1.877	-1.852	-1.571
	.270	-1.182	-1.482	-1.498	-1.358	-1.186
	.400	-1.014	-1.142	-1.132	-.994	-.962
	.620	-.975	-1.249	-.693	-.526	-.500
	.685	-2.729	-5.329	-.778	-1.631	-2.321
	.693	-2.365	-5.856	-2.054	-1.813	-1.936
	.700	-1.455	-4.153	-1.805	-1.455	-1.750
	.720	-.741	-1.696	-1.302	-1.156	-1.282
	.750	-.637	-.995	-1.132	-1.111	-1.154
	.800	-.552	-.681	-.948	-.897	-1.013
	.900	-.468	-.694	-.870	-.851	-.891
	.980	-.422	-.528	-.831	-.845	-.897
Lower	.025	.468	.708	.680	.702	.558
	.120	.819	.841	.739	.741	.603
	.220	.780	.781	.739	.721	.615
	.300	.676	.694	.667	.650	.526
	.620	.754	.801	.733	.708	.282
	.750	.845	.895	.739	.708	.622
	.850	.624	.674	.556	.552	.468
	.950	.377	.387	.170	.175	.122

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TABLE 19 Concluded
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 2.0 h_d/c = 1.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = 18.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.183	.756	-.288	.147
.053	-.314	.557	-.425	-.053
.100	-.209	.318	-.491	-.287
.145	-.170	.225	-.464	-.321
.189	-.105	.252	-.412	-.260
.234	-.105	.292	-.092	-.307
.280	-.131	.298	.111	-.347
.326	-.170	.325	.052	-.441
.371	-.347	.418	-.360	-.487
.392	-.430	.480	-.844	-.601
.413	-.504	.544	-1.112	.160
.434	-.530	.570	-1.727	.621
.457	-.523	.560	-1.380	.694
.480	-.517	.550	-1.105	.654
.502	-.543	.540	-.994	.621
.551	-.432	.525	-.981	.594
.585	-.399	.517	-1.099	.668
.592	-.373	.504	-1.204	-1.142
.613	-.242	.405	-.909	.550
.634	-.235	.265	-.615	-.447
.655	-.203	.146	-.366	-.414
.675	-.118	.027	-.203	-.254
.696	-.078	-.073	-.092	-.067
.774	-.007	.093	.039	.100
.852	-.105	.099	-.052	-.027
.930	-.026	.106	-.026	.107

Surface	x/c	Wing, flap, or aileron 0.221	0.423	0.640	0.800	0.918
Upper	.010	-4.472	-2.991	-3.345	-3.172	-2.845
	.080	-1.785	-2.029	-3.018	-2.957	-1.334
	.130	-3.227	-3.077	-3.218	-3.002	-2.832
	.145	-9.932	-9.098	-8.460	-8.176	-7.594
	.155	-4.268	-4.489	-4.514	-4.082	-3.146
	.180	-2.852	-2.984	-3.172	-2.800	-2.433
	.220	-1.772	-2.122	-2.257	-2.047	-1.550
	.270	-1.324	-1.684	-1.796	-1.570	-1.164
	.400	-1.014	-1.285	-1.309	-1.145	-1.262
	.620	-.988	-1.419	-1.028	-.883	-1.073
	.685	-3.016	-4.467	-.928	-1.930	-1.949
	.693	-2.839	-4.934	-2.317	-2.132	-1.655
	.700	-1.778	-3.523	-2.017	-1.655	-1.387
	.720	-.843	-1.512	-1.436	-1.302	-.942
	.750	-.645	-.909	-1.235	-1.112	-.981
	.800	-.566	-.584	-1.028	-.975	-.968
	.900	-.461	-.477	-.921	-.935	-.962
	.980	-.415	-.285	-.868	-.896	-.903
Lower	.025	.685	.849	.828	.818	.687
	.120	.869	.855	.775	.733	.608
	.220	.823	.809	.801	.759	.654
	.300	.738	.755	.728	.693	.595
	.620	.777	.822	.768	.713	.288
	.750	.843	.875	.788	.733	.621
	.850	.672	.695	.588	.582	.471
	.950	.448	.444	.194	.216	.144

α = 22.9°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.231	.823	-.446	.033
.053	-.342	.626	-.541	-.105
.100	-.211	.402	-.586	-.369
.145	-.171	.303	-.548	-.421
.189	-.092	.323	-.509	-.356
.234	-.066	.356	-.108	-.435
.280	-.112	.369	.140	-.454
.326	-.198	.375	.076	-.580
.371	-.415	.487	-.414	-.645
.392	-.525	.440	-.968	-.856
.413	-.606	.599	-1.248	.191
.434	-.632	.626	-1.993	.665
.457	-.553	.610	-1.547	.731
.480	-.507	.600	-1.312	.705
.502	-.481	.580	-1.146	.659
.551	-.316	.560	-1.044	.612
.585	-.263	.540	-1.146	.665
.592	-.244	.520	-1.541	-1.462
.613	-.119	.415	-1.216	-.211
.634	-.158	.283	-.726	-.700
.655	-.145	.158	-.408	-.395
.675	-.066	.033	-.204	-.171
.696	-.026	-.007	-.089	-.020
.774	-.040	.060	.032	.105
.852	-.125	.119	-.076	.013
.930	-.040	.132	-.025	.132

Surface	x/c	Wing, flap, or aileron 0.221	0.423	0.640	0.800	0.918
Upper	.010	-8.725	-3.933	-4.268	-3.859	-3.688
	.080	-2.039	-3.683	-4.320	-3.891	-2.338
	.130	-3.323	-3.003	-3.385	-3.267	-2.529
	.145	-9.622	-8.213	-7.100	-6.412	-6.691
	.155	-4.072	-4.393	-4.162	-3.579	-2.885
	.180	-2.685	-3.053	-3.095	-2.630	-2.298
	.220	-1.665	-2.213	-2.239	-1.942	-1.475
	.270	-1.400	-1.737	-1.778	-1.541	-1.120
	.400	-1.045	-1.317	-1.324	-1.172	-1.080
	.620	-.974	-1.353	-1.179	-.866	-1.060
	.685	-3.814	-3.589	-.724	-1.738	-1.936
	.693	-3.814	-3.974	-2.075	-1.923	-1.673
	.700	-2.510	-2.911	-1.811	-1.407	-1.436
	.720	-1.284	-1.254	-1.185	-1.044	-.962
	.750	-.865	-.784	-1.047	-.968	-.968
	.800	-.594	-.533	-.975	-.904	-.968
	.900	-.432	-.393	-.836	-.866	-.948
	.980	-.419	-.244	-.764	-.866	-.869
Lower	.025	.800	.883	.836	.815	.692
	.120	.910	.883	.777	.764	.593
	.220	.858	.823	.817	.771	.659
	.300	.787	.797	.744	.713	.606
	.620	.800	.843	.771	.720	.277
	.750	.865	.883	.790	.739	.593
	.850	.697	.731	.626	.592	.494
	.950	.484	.487	.250	.242	.191

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TABLE 20 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = -1.4°$											
.032	.285	.296	.280	.295		.010	.955	.888	.848	.828	.810
.053	.063	.099	.061	.069		.080	.475	.401	.346	.335	.354
.100	-.082	-.025	-.103	-.082		.130	-.331	-.505	-.591	-.499	-.557
.145	-.082	-.074	-.067	-.050		.145	-4.327	-4.006	-3.658	-3.883	-3.701
.189	-.013	-.006	-.012	-.006		.155	-1.561	-1.535	-1.433	-1.290	-1.063
.234	-.044	.068	-.012	.013		.180	-1.180	-.968	-.999	-.828	-1.018
.280	-.038	.080	-.049	.031		.220	-.687	-.684	-.654	-.676	-.633
.326	-.038	.105	-.049	.082		.270	-.543	-.586	-.515	-.414	-.468
.371	-.082	.136	-.073	.151	Upper	.400	-.562	-.555	-.302	-.310	-.449
.392	-.032	.148	.006	.346		.620	-1.074	-.875	.547	-.706	-.841
.413	-.139	.160	-.128	-.151		.685	-6.056	-4.278	.126	-1.193	-6.756
.434	-.183	.160	-.341	-.264		.693	-5.988	-4.684	-1.238	-1.473	-5.877
.457	-.209	.180	-.438	-.189		.700	-3.752	-3.328	-1.106	-1.047	-4.137
.480	-.278	.210	-.396	-.025		.720	-1.792	-1.208	-.760	-.742	-1.290
.502	-.354	.230	-.426	.107		.750	-1.199	-.733	-.786	-.779	-.727
.551	-.380	.260	-.572	.289		.800	-.799	-.647	-.792	-.767	-.639
.585	-.386	.296	-.712	.251		.900	-.537	-.616	-.704	-.803	-.620
.592	-.373	.296	-.822	-.855		.980	-.100	-.542	-.710	-.706	-.386
.613	-.291	.210	-.676	-.666		.025	-.312	-.173	.019	.079	-.038
.634	-.247	.160	-.475	-.723		.120	-.350	-.136	.000	.012	-.057
.655	-.209	.105	-.341	-.176		.220	-.312	-.173	-.038	.000	-.051
.675	-.120	.012	-.237	-.101	Lower	.300	-.087	-.222	-.094	-.049	-.076
.696	-.076	.000	-.176	-.050		.620	.543	.357	.050	-.024	-.190
.774	.013	.049	-.049	.038		.750	.824	.598	.088	.037	.101
.852	.013	-.031	.012	-.094		.850	.699	.647	.289	.134	.215
.930	.082	-.160	.097	-.201		.950	.524	.357	.157	.079	.240
$\alpha = 5.8°$											
.032	.082	.506	.147	.337		.010	.647	.679	.624	.667	.639
.053	-.095	.263	-.077	.112		.080	-.115	-.122	-.137	-.167	-.108
.100	-.190	.083	-.199	-.087		.130	-1.224	-1.385	-1.374	-1.333	-1.368
.145	-.139	.006	-.167	-.056		.145	-6.442	-6.045	-5.201	-5.891	-5.402
.189	-.057	.064	-.103	-.019		.155	-2.615	-2.558	-2.316	-2.295	-1.936
.234	-.095	.109	-.032	-.012		.180	-1.885	-1.628	-1.548	-1.462	-1.607
.280	-.095	.135	.038	-.019		.220	-1.154	-1.122	-1.036	-1.128	-1.063
.326	-.089	.141	.013			.270	-.885	-.917	-.787	-.744	-.810
.371	-.190	.212	-.128	.037	Upper	.400	-.737	-.744	-.431	-.481	-.734
.392	-.215	.250	-.224	.206		.620	-1.128	-.942	.425	-.718	-1.075
.413	-.266	.282	-.519	.150		.685	-5.660	-4.385	.131	-1.410	-8.129
.434	-.348	.321	-.795	.169		.693	-5.333	-4.436	-1.136	-1.667	-7.363
.457	-.354	.340	-.731	.312		.700	-3.301	-3.128	-1.018	-1.199	-5.478
.480	-.418	.360	-.667	.393		.720	-1.583	-1.071	-.681	-.782	-2.062
.502	-.481	.380	-.622	.425		.750	-1.122	-.679	-.706	-.821	-1.347
.551	-.493	.400	-.705	.468		.800	-.827	-.724	-.762	-.821	-1.031
.585	-.462	.417	-.833	.500		.900	-.571	-.641	-.656	-.846	-.822
.592	-.443	.417	-.917	-.887		.980	-.167	-.590	-.624	-.718	-.367
.613	-.335	.333	-.712	-.612		.025	.038	.365	.343	.346	.171
.634	-.304	.231	-.474	-.481		.120	.154	.301	.300	.256	.070
.655	-.259	.147	-.321	-.125		.220	.551	.295	.281	.237	.089
.675	-.177	.006	-.231	-.062	Lower	.300	.609	.494	.437	.378	.361
.696	-.120	-.006	-.160	-.044		.620	.705	.699	.649	.641	.310
.774	.000	.071	-.058	.037		.750	.808	.776	.687	.699	.620
.852	-.013	.026	.013	-.087		.850	.628	.583	.556	.571	.557
.930	.051	-.122	.090	-.150		.950	.449	.224	.175	.192	.367
$\alpha = 13.3°$											
.032	-.058	.643	-.076	.290		.010	-.083	-1.020	-1.106	-1.121	-.303
.053	-.232	.435	-.255	.059		.080	-.897	-.916	-.955	-.993	-.800
.100	-.187	.201	-.350	-.198		.130	-2.269	-2.475	-2.568	-2.388	-2.420
.145	-.129	.136	-.306	-.165		.145	-8.276	-8.082	-7.343	-7.667	-7.286
.189	-.084	.162	-.242	-.139		.155	-3.462	-3.645	-3.504	-3.318	-2.827
.234	-.084	.208	-.064	-.171		.180	-2.333	-2.306	-2.325	-2.101	-2.226
.280	-.110	.227	.089	-.171		.220	-1.462	-1.605	-1.541	-1.522	-1.452
.326	-.123	.234	.064	-.211		.270	-1.083	-1.254	-1.159	-1.044	-1.097
.371	-.245	.318	-.185	-.211	Upper	.400	-.891	-.916	-.619	-.605	-.923
.392	-.300	.370	-.560	-.171		.620	-1.122	-1.046	.040	-.751	-1.284
.413	-.361	.435	-.891	.204		.685	-4.205	-3.905	-.290	-1.853	-8.918
.434	-.419	.455	-1.172	.527		.693	-3.788	-3.729	-1.218	-1.458	-8.112
.457	-.419	.460	-.974	.612		.700	-2.449	-2.592	-1.100	-1.025	-6.027
.480	-.445	.465	-.815	.580		.720	-1.353	-.838	-.718	-.694	-2.304
.502	-.510	.473	-.751	.540		.750	-1.013	-.734	-.738	-.726	-1.458
.551	-.458	.480	-.847	.566		.800	-.724	-.747	-.751	-.745	-1.065
.585	-.445	.487	-.885	.632		.900	-.385	-.708	-.738	-.771	-.878
.592	-.407	.474	-.942	-.935		.980	-.205	-.669	-.659	-.662	-.471
.613	-.297	.383	-.713	-.487		.025	.462	.695	.672	.662	.523
.634	-.277	.273	-.452	-.540		.120	.801	.812	.757	.726	.587
.655	-.245	.156	-.293	-.086		.220	.750	.747	.751	.726	.620
.675	-.161	.019	-.191	-.040	Lower	.300	.647	.689	.659	.630	.523
.696	-.123	.019	-.134	-.013		.620	.724	.799	.751	.675	.219
.774	.006	.123	-.013	.066		.750	.833	.864	.751	.720	.613
.852	-.032	.084	-.013	-.040		.850	.635	.630	.540	.560	.510
.930	.045	-.026	.051	-.040		.950	.442	.266	.178	.204	.297

TABLE 20 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = 19.0°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.151	.758	-.303	.145
.053	-.283	.548	-.435	-.046
.100	-.184	.332	-.487	-.277
.145	-.138	.217	-.448	-.283
.189	-.079	.257	-.395	-.250
.234	-.066	.298	-.105	-.290
.280	-.105	.311	.105	-.310
.326	-.132	.345	.013	-.395
.371	-.303	.433	-.283	-.415
.392	-.375	.490	-.790	-.520
.413	-.454	.555	-1.067	.184
.434	-.501	.582	-1.594	.619
.457	-.487	.570	-1.304	.692
.480	-.474	.560	-1.008	.659
.502	-.494	.550	-.922	.606
.551	-.389	.538	-.948	.593
.585	-.356	.528	-1.041	.645
.592	-.329	.521	-1.251	-1.218
.613	-.224	.406	-1.041	-.800
.634	-.211	.278	-.626	-.619
.655	-.191	.142	-.342	-.290
.675	-.105	.014	-.198	-.066
.696	-.066	.047	-.105	-.033
.774	.007	.196	.000	.092
.852	-.046	.108	-.059	.000
.930	.000	.095	.013	.086

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-3.999	-2.885	-2.970	-2.904	-2.674
	.080	-1.691	-1.950	-2.536	-2.509	-1.475
	.130	-3.090	-2.973	-2.858	-2.792	-3.036
	.145	-9.556	-8.884	-7.673	-8.101	-8.239
	.155	-4.085	-4.347	-3.978	-3.919	-3.556
	.180	-2.732	-2.851	-2.727	-2.615	-2.759
	.220	-1.664	-2.004	-1.870	-1.870	-1.897
	.270	-1.227	-1.564	-1.383	-1.344	-1.482
	.400	-.935	-1.083	-.705	-.771	-1.232
	.620	-1.074	-1.144	-.217	-.836	-1.350
	.685	-3.906	-2.973	-.040	-2.246	-7.653
	.693	-3.707	-2.661	-1.245	-2.338	-6.823
	.700	-2.394	-1.950	-1.113	-1.752	-4.959
	.720	-1.260	-.860	-.738	-.982	-1.726
	.750	-.935	-.907	-.751	-.797	-1.014
	.800	-.710	-.894	-.757	-.777	-.738
	.900	-.491	-.887	-.751	-.817	-.803
	.980	-.365	-.826	-.698	-.744	-.606
Lower	.025	.716	.846	.817	.790	.645
	.120	.882	.853	.764	.731	.580
	.220	.809	.813	.790	.751	.612
	.300	.749	.758	.718	.678	.553
	.620	.789	.840	.764	.692	.231
	.750	.855	.887	.744	.698	.606
	.850	.670	.677	.566	.573	.501
	.950	.477	.298	.224	.250	.290

$\alpha = 23.0°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.256	.790	-.430	.066
.053	-.350	.606	-.538	-.086
.100	-.195	.395	-.612	-.369
.145	-.168	.290	-.565	-.395
.189	-.067	.342	-.498	-.329
.234	-.074	.356	-.155	-.421
.280	-.114	.375	.141	-.435
.326	-.188	.382	.013	-.547
.371	-.410	.474	-.430	-.645
.392	-.500	.540	-.968	-.823
.413	-.605	.586	-1.244	.198
.434	-.632	.619	-2.004	.672
.457	-.558	.600	-1.493	.731
.480	-.498	.580	-1.264	.705
.502	-.498	.560	-1.096	.672
.551	-.336	.550	-1.029	.606
.585	-.289	.547	-1.116	.645
.592	-.256	.514	-1.520	-1.416
.613	-.168	.408	-1.304	-.750
.634	-.161	.283	-.780	-.659
.655	-.155	.145	-.430	-.290
.675	-.081	.026	-.269	-.119
.696	-.054	.040	-.168	-.053
.774	-.034	.204	.000	.079
.852	-.121	.119	-.074	-.013
.930	-.047	.145	-.034	.132

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-7.934	-3.728	-3.833	-3.712	-3.429
	.080	-1.943	-3.438	-3.905	-3.718	-1.997
	.130	-3.225	-2.819	-3.082	-2.918	-2.347
	.145	-9.517	-7.890	-6.217	-6.166	-6.233
	.155	-4.010	-4.136	-3.596	-3.281	-2.522
	.180	-2.623	-2.819	-2.568	-2.320	-2.017
	.220	-1.596	-2.009	-1.791	-1.647	-1.244
	.270	-1.282	-1.541	-1.330	-1.231	-.928
	.400	-.962	-1.080	-.929	-.867	-.867
	.620	-.948	-.955	-.738	-.968	-.867
	.685	-3.853	-1.765	-.408	-2.071	-1.708
	.693	-3.944	-1.936	-1.594	-2.205	-1.392
	.700	-2.603	-1.502	-1.416	-1.385	-1.204
	.720	-1.367	-.817	-1.027	-.894	-.807
	.750	-.975	-.810	-.935	-.861	-.834
	.800	-.693	-.803	-.810	-.807	-.814
	.900	-.458	-.784	-.764	-.773	-.814
	.980	-.327	-.724	-.738	-.740	-.753
Lower	.025	.785	.869	.856	.854	.706
	.120	.896	.863	.777	.800	.619
	.220	.850	.817	.803	.793	.679
	.300	.778	.803	.757	.753	.625
	.620	.778	.836	.771	.740	.323
	.750	.857	.869	.790	.740	.639
	.850	.680	.698	.593	.612	.524
	.950	.477	.362	.237	.269	.222

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TABLE 20 (b) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 4.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.5°											
.032	.272	.308	.288	.318	Upper	.010	.962	.875	.868	.833	.848
.053	.063	.086	.061	.062		.080	.456	.382	.325	.300	.373
.100	-.089	-.043	-.098	-.069		.130	-.368	-.561	-.624	-.576	-.582
.145	-.089	-.049	-.067	-.037		.145	-4.433	-4.130	-3.771	-4.128	-3.808
.189	-.006	[illegible]	-.018	.012		.155	-1.630	-1.603	-1.548	-1.445	-1.145
.234	-.044	.062	-.006	.019		.180	-1.218	-1.023	-1.068	-.919	-1.050
.280	-.051	.086	-.055	.037		.220	-.718	-.746	-.724	-.772	-.671
.326	-.019	.105	-.037	.075		.270	-.581	-.647	-.599	-.508	-.487
.371	-.101	.142	-.067	.144		.400	-.587	-.629	-.437	-.447	-.512
.392	-.038	.155	.006	.343		.620	-1.099	-.986	.150	-.704	-.848
.413	-.152	.179	-.147	-.162		.685	-6.106	-4.635	.031	-1.678	-6.427
.434	-.202	.179	-.361	-.300		.693	-6.038	-5.067	-1.224	-1.899	-5.491
.457	-.234	.200	-.453	-.200		.700	-3.809	-3.556	-1.118	-1.298	-3.846
.480	-.316	.220	-.435	-.019		.720	-1.842	-1.405	-.799	-.784	-1.234
.502	-.373	.240	-.453	.125		.750	-1.236	-.807	-.830	-.790	-.778
.551	-.418	.260	-.619	.312		.800	-.812	-.610	-.830	-.808	-.702
.585	-.399	.284	-.747	.275		.900	-.524	-.641	-.712	-.790	-.664
.592	-.373	.290	-.870	-.899		.980	-.050	-.561	-.706	-.674	-.474
.613	-.304	.222	-.723	-.850							
.634	-.247	.173	-.502	-.799	Lower	.025	-.256	-.111	.012	.104	.006
.655	-.183	.117	-.343	-.175		.120	-.287	-.086	[illegible]	.049	-.051
.675	-.127	.018	-.245	-.075		.220	-.237	-.123	-.031	.024	-.006
.696	-.063	.012	-.171	-.025		.300	-.025	-.148	-.087	-.012	-.038
.774	-.013	.062	-.067	.050		.620	.556	.339	.037	-.012	-.202
.852	-.013	-.037	.018	-.106		.750	.812	.555	.106	.061	.057
.930	.063	-.154	.092	-.200		.850	.712	.647	.287	.171	.139
						.950	.537	.401	.175	.147	.164
α = 5.8°											
.032	.097	.494	.138	.307	Upper	.010	.618	.628	.582	.612	.632
.053	-.103	.269	-.079	.092		.080	-.100	-.154	-.203	-.257	-.142
.100	-.187	.064	-.217	-.098		.130	-1.224	-1.417	-1.531	-1.515	-1.458
.145	-.129	.013	-.184	-.072		.145	-6.300	-6.096	-5.651	-6.336	-5.743
.189	-.065	.064	-.105	-.033		.155	-2.547	-2.596	-2.558	-2.542	-2.065
.234	-.090	.135	-.026	-.033		.180	-1.854	-1.647	-1.753	-1.646	-1.717
.280	-.103	.128	.046	-.020		.220	-1.149	-1.186	-1.210	-1.291	-1.149
.326	-.103	.154	.020	-.020		.270	-.862	-.974	-.975	-.909	-.884
.371	-.181	.218	-.138	.033		.400	-.749	-.814	-.674	-.711	-.826
.392	-.230	.250	-.244	.177		.620	-1.130	-1.090	-.020	-.784	-1.162
.413	-.284	.295	-.533	.137		.685	-5.401	-4.660	.085	-2.140	-8.454
.434	-.348	.321	-.830	.203		.693	-5.195	-5.006	-1.249	-2.259	-7.615
.457	-.368	.340	-.771	.360		.700	-3.247	-3.564	-1.105	-1.640	-5.672
.480	-.432	.360	-.692	.445		.720	-1.573	-1.321	-.791	-.968	-2.226
.502	-.490	.380	-.652	.438		.750	-1.143	-.833	-.844	-.915	-1.484
.551	-.490	.400	-.764	.491		.800	-.837	-.731	-.870	-.909	-1.168
.585	-.465	.423	-.889	.504		.900	-.524	-.718	-.680	-.889	-.974
.592	-.426	.423	-.968	-.962		.980	-.131	-.654	-.661	-.724	-.471
.613	-.336	.340	-.731	-.602							
.634	-.277	.256	-.501	-.530	Lower	.025	.081	.372	.353	.375	.194
.655	-.239	.141	-.342	-.092		.120	.181	.308	.301	.277	.097
.675	-.136	.026	-.224	-.026		.220	.524	.333	.301	.250	.116
.696	-.097	.026	-.138	-.026		.300	.612	.487	.471	.435	.445
.774	-.006	.077	-.020	.046		.620	.687	.705	.667	.665	.310
.852	-.013	.051	.020	-.085		.750	.799	.769	.700	.705	.613
.930	.090	-.115	.092	-.131		.850	.624	.590	.543	.566	.549
						.950	.456	.263	.157	.178	.323
α = 13.3°											
.032	-.087	.637	-.079	.262	Upper	.010	-.162	-1.150	-1.347	-1.297	-.274
.053	-.254	.429	-.250	.039		.080	-.929	-.949	-1.053	-1.113	-.855
.100	-.187	.208	-.356	-.177		.130	-2.326	-2.547	-2.721	-2.601	-2.557
.145	-.134	.136	-.316	-.183		.145	-8.348	-8.186	-7.607	-8.213	-7.686
.189	-.073	.156	-.270	-.157		.155	-3.508	-3.729	-3.696	-3.649	-3.092
.234	-.107	.221	-.072	-.177		.180	-2.358	-2.397	-2.486	-2.345	-2.390
.280	-.120	.234	.066	-.177		.220	-1.481	-1.689	-1.714	-1.739	-1.596
.326	-.127	.260	.013	-.229		.270	-1.117	-1.345	-1.308	-1.251	-1.242
.371	-.254	.325	-.211	-.222		.400	-.942	-.988	-.844	-.850	-1.075
.392	-.315	.375	-.586	-.183		.620	-1.072	-1.169	-.340	-.909	-1.482
.413	-.374	.435	-.929	.183		.685	-3.391	-4.457	-.412	-3.339	-9.535
.434	-.441	.474	-1.218	.523		.693	-2.716	-4.379	-1.204	-2.470	-8.620
.457	-.427	.476	-1.027	.615		.700	-1.696	-3.073	-1.112	-1.870	-6.424
.480	-.467	.478	-.883	.563		.720	-.916	-1.065	-.759	-1.021	-2.551
.502	-.534	.480	-.803	.523		.750	-.767	-.695	-.772	-.929	-1.669
.551	-.467	.483	-.856	.549		.800	-.650	-.773	-.818	-.876	-1.269
.585	-.441	.487	-.863	.621		.900	-.487	-.786	-.752	-.876	-1.088
.592	-.407	.468	-.869	-.883		.980	-.422	-.754	-.680	-.718	-.621
.613	-.294	.377	-.711	-.600							
.634	-.280	.260	-.514	-.523	Lower	.025	.487	.682	.674	.659	.534
.655	-.240	.136	-.342	-.379		.120	.799	.793	.746	.718	.588
.675	-.154	.019	-.198	-.157		.220	.754	.728	.746	.711	.628
.696	-.114	.039	-.125	-.020		.300	.656	.676	.667	.626	.507
.774	[illegible]	.156	.020	.092		.620	.734	.780	.746	.678	.207
.852	-.053	.091	-.026	-.059		.750	.812	.871	.739	.711	.601
.930	.040	[illegible]	.026	-.013		.850	.630	.630	.530	.553	.481
						.950	.390	.260	.170	.217	.254

TABLE 20 (b) Continued / Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = 19.0°$											
.032	-.162	.758	-.318	.160	Upper	.010	-4.013	-2.912	-3.212	-3.103	-2.820
.053	-.299	.569	-.458	-.033		.080	-1.736	-1.889	-2.831	-2.838	-1.598
.100	-.201	.311	-.511	-.300		.130	-3.172	-3.020	-3.032	-2.865	-3.131
.145	-.156	.210	-.471	-.300		.145	-9.762	-8.972	-8.040	-7.858	-8.472
.189	-.084	.230	-.424	-.267		.155	-4.187	-4.408	-4.247	-3.906	-3.723
.234	-.091	.311	-.139	-.327		.180	-2.758	-2.925	-2.951	-2.639	-2.911
.280	-.117	.318	.099	-.327		.220	-1.709	-2.059	-2.063	-1.916	-2.021
.326	-.143	.332	.033	-.421		.270	-1.255	-1.591	-1.589	-1.426	-1.631
.371	-.312	.413	-.298	-.474		.400	-.968	-1.151	-.988	-.942	-1.358
.392	-.375	.470	-.855	-.561		.620	-1.028	-1.232	-.441	-.935	-1.455
.413	-.461	.542	-1.107	.187		.685	-3.319	-1.569	.007	-2.971	-7.796
.434	-.520	.569	-1.737	.634		.693	-3.172	-1.636	-1.342	-3.349	-6.906
.457	-.507	.560	-1.346	.708		.700	-2.017	-2.553	-1.229	-2.089	-5.100
.480	-.487	.550	-1.068	.668		.720	-1.022	-1.022	-.855	-1.081	-1.936
.502	-.507	.540	-.981	.621		.750	-.781	-.772	-.855	-1.068	-1.241
.551	-.403	.530	-.968	.608		.800	-.648	-.853	-.875	-.975	-.962
.585	-.377	.528	-1.028	.668		.900	-.534	-.867	-.881	-.942	-.897
.592	-.357	.515	-1.253	-1.169		.980	-.487	-.826	-.775	-.869	-.637
.613	-.240	.393	-1.041	-.650	Lower	.025	.688	.840	.828	.809	.650
.634	-.240	.257	-.690	-.541		.120	.875	.840	.761	.723	.546
.655	-.195	.135	-.411	-.421		.220	.821	.792	.815	.723	.617
.675	-.130	.020	-.239	-.160		.300	.741	.745	.714	.723	.546
.696	-.091	.020	-.126	-.007		.620	.761	.819	.781	.670	.234
.774	.019	.203	.020	.120		.750	.848	.880	.755	.703	.591
.852	-.071	.129	-.080	-.020		.850	.661	.677	.581	.570	.481
.930	.	.122	-.020	.107		.950	.434	.345	.214	.206	.253
$\alpha = 23.0°$											
.032	-.225	.865	-.467	.080	Upper	.010	-8.681	-4.109	-4.093	-3.997	-3.601
.053	-.345	.678	-.584	-.087		.080	-2.071	-1.929	-4.167	-4.052	-2.288
.100	-.199	.429	-.673	-.354		.130	-3.396	-1.071	-3.299	-3.249	-2.420
.145	-.179	.318	-.632	-.381		.145	-9.905	-3.163	-6.677	-6.655	-6.399
.189	-.080	.339	-.549	-.327		.155	-4.189	-4.434	-3.906	-3.654	-2.752
.234	-.060	.380	-.165	-.414		.180	-2.757	-1.085	-2.878	-2.624	-2.149
.280	-.126	.387	.117	-.414		.220	-1.688	-2.234	-2.050	-1.923	-1.373
.326	-.179	.401	.021	-.534		.270	-1.412	-1.757	-1.596	-1.477	-1.041
.371	-.438	.512	-.467	-.608		.400	-1.056	-1.300	-1.142	-1.120	-.968
.392	-.520	.550	-1.044	-.815		.620	-.995	-1.273	-.942	-1.106	-.955
.413	-.603	.602	-1.319	.220		.685	-3.947	-2.871	-.434	-1.999	-1.797
.434	-.637	.643	-2.115	.694		.693	-3.960	-1.196	-1.756	-2.157	-1.466
.457	-.570	.625	-1.593	.755		.700	-2.602	-2.290	-1.522	-1.449	-1.293
.480	-.504	.610	-1.346	.714		.720	-1.338	-1.024	-1.102	-.996	-.915
.502	-.497	.595	-1.174	.681		.750	-.908	-.782	-1.008	-.975	-.942
.551	-.318	.570	-1.106	.641		.800	-.652	-.733	-.868	-.934	-.922
.585	-.285	.567	-1.154	.674		.900	-.504	-.692	-.848	-.879	-.889
.592	-.252	.546	-1.518	-1.456		.980	-.430	-.546	-.801	-.859	-.842
.613	-.166	.443	-1.374	-.820	Lower	.025	.800	.913	.861	.838	.676
.634	-.146	.304	-.824	-.528		.120	.914	.879	.788	.783	.590
.655	-.139	.159	-.460	-.367		.220	.867	.872	.835	.804	.676
.675	-.066	.035	-.275	-.114		.300	.793	.816	.755	.762	.590
.696	-.040	.069	-.124	.020		.620	.800	.858	.781	.714	.285
.774	-.020	.221	.007	.016		.750	.874	.913	.788	.735	.603
.852	-.113	.118	-.096	.013		.850	.672	.719	.634	.591	.484
.930	-.027	.152	-.041	.140		.950	.471	.443	.234	.234	.179

TABLE 20 (c) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = -1.5°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.267	.300	.287	.275
.053	.019	.096	.062	.050
.100	-.108	-.036	-.094	-.081
.145	-.115	-.060	-.075	-.062
.189	-.045	-.018	[illegible]	-.031
.234	-.057	.048	.006	.006
.280	-.070	.084	-.025	.025
.326	-.057	.084	-.025	.062
.371	-.108	.132	-.087	.137
.392	-.032	.145	.025	.331
.413	-.172	.168	-.137	-.094
.434	-.197	.168	-.381	-.244
.457	-.274	.200	-.500	-.150
.480	-.312	.230	-.456	-.006
.502	-.414	.260	-.481	.119
.551	-.439	.290	-.662	.312
.585	-.414	.300	-.787	.268
.592	-.388	.300	-.930	-.949
.613	-.312	.240	-.768	-.830
.634	-.261	.198	-.537	-.812
.655	-.204	.114	-.368	-.187
.675	-.121	.036	-.250	-.069
.696	-.057	.030	-.162	-.031
.774	-.032	.078	-.056	.012
.852	.006	-.036	.012	-.119
.930	.083	-.168	.100	-.212

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.962	.883	.843	.824	.809
	.080	.459	.337	.262	.262	.299
	.130	-.355	-.613	-.724	-.699	-.681
	.145	-4.367	-4.195	-4.021	-4.508	-4.145
	.155	-1.611	-1.671	-1.686	-1.623	-1.293
	.180	-1.194	-1.082	-1.193	-1.055	-1.204
	.220	-.710	-.799	-.849	-.924	-.796
	.270	-.570	-.709	-.743	-.662	-.624
	.400	-.619	-.691	-.656	-.656	-.643
	.620	-1.121	-1.136	-.368	-.849	-1.000
	.685	-6.204	-5.216	-.237	-2.522	-6.056
	.693	-6.137	-5.727	-1.586	-2.654	-5.088
	.700	-3.895	-4.050	-1.380	-1.655	-3.617
	.720	-1.892	-1.659	-1.030	-.937	-1.280
	.750	-1.274	-.950	-.912	-.968	-.930
	.800	-.839	-.613	-.662	-.887	-.828
	.900	-.527	-.487	-.649	-.837	-.745
	.980	-.018	-.397	-.643	-.768	-.560
Lower	.025	-.220	-.078	.056	.119	-.032
	.120	-.276	-.048	.037	.050	-.051
	.220	-.202	-.072	.006	.025	-.057
	.300	.012	-.132	-.056	-.044	-.083
	.620	.551	.373	.112	.062	-.159
	.750	.802	.589	.175	.156	.096
	.850	.717	.661	.356	.268	.236
	.950	.551	.469	.212	.212	.210

$\alpha = 5.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.094	.501	.145	.310
.053	-.082	.274	-.066	.072
.100	-.182	.073	-.211	-.125
.145	-.113	.007	-.171	-.099
.189	-.057	.060	-.119	-.033
.234	-.082	.114	-.020	-.040
.280	-.094	.120	.059	-.053
.326	-.088	.140	.020	-.026
.371	-.170	.220	-.158	.020
.392	-.220	.255	-.277	.165
.413	-.270	.294	-.566	.158
.434	-.352	.347	-.883	.217
.457	-.365	.365	-.810	.362
.480	-.427	.395	-.724	.461
.502	-.509	.425	-.705	.454
.551	-.484	.455	-.777	.494
.585	-.446	.461	-.889	.520
.592	-.434	.461	-.922	-.883
.613	-.327	.367	-.724	-.700
.634	-.258	.280	-.514	-.659
.655	-.220	.174	-.342	-.125
.675	-.132	.060	-.211	-.046
.696	-.094	.047	-.112	-.007
.774	-.019	.087	-.020	.040
.852	-.019	.073	-.007	-.099
.930	.044	-.093	.079	-.165

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.574	.581	.553	.580	.622
	.080	-.181	-.274	-.257	-.329	-.157
	.130	-1.329	-1.623	-1.653	-1.620	-1.521
	.145	-6.653	-6.644	-6.013	-6.711	-5.857
	.155	-2.710	-2.911	-2.786	-2.753	-2.162
	.180	-1.988	-1.890	-1.903	-1.805	-1.779
	.220	-1.239	-1.356	-1.357	-1.436	-1.213
	.270	-.955	-1.175	-1.139	-1.067	-.962
	.400	-.826	-1.008	-.948	-.922	-.905
	.620	-1.129	-1.342	-.514	-1.001	-1.269
	.685	-4.440	-5.589	-.296	-2.147	-8.622
	.693	-4.066	-6.043	-1.732	-2.298	-7.843
	.700	-2.375	-4.320	-1.521	-1.455	-5.863
	.720	-1.116	-1.783	-1.153	-1.027	-2.451
	.750	-.942	-1.062	-1.060	-1.106	-1.691
	.800	-.852	-.781	-.705	-1.001	-1.351
	.900	-.600	-.795	-.757	-.896	-1.125
	.980	-.413	-.728	-.692	-.902	-.584
Lower	.025	.090	.387	.362	.421	.207
	.120	.155	.327	.316	.283	.101
	.220	.510	.414	.316	.329	.157
	.300	.587	.561	.507	.540	.496
	.620	.684	.714	.659	.665	.302
	.750	.807	.775	.705	.698	.610
	.850	.581	.608	.547	.560	.509
	.950	.394	.314	.165	.184	.283

$\alpha = 13.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.092	.652	-.097	.275
.053	-.242	.448	-.273	.059
.100	-.183	.231	-.338	-.190
.145	-.150	.138	-.305	-.164
.189	-.092	.171	-.247	-.118
.234	-.105	.224	-.078	-.144
.280	-.124	.224	.078	-.157
.326	-.144	.244	.052	-.216
.371	-.275	.329	-.221	-.216
.392	-.325	.380	-.604	-.190
.413	-.379	.428	-.929	.203
.434	-.432	.481	-1.215	.536
.457	-.445	.482	-1.040	.608
.480	-.477	.483	-.897	.576
.502	-.530	.484	-.812	.543
.551	-.464	.485	-.871	.569
.585	-.438	.487	-.845	.634
.592	-.406	.481	-.858	-.857
.613	-.307	.369	-.689	-.550
.634	-.281	.257	-.494	-.445
.655	-.249	.138	-.325	-.399
.675	-.144	[illegible]	-.169	-.203
.696	-.092	.046	-.091	-.033
.774	[illegible]	.165	.045	.092
.852	-.052	.086	-.013	-.033
.930	.033	[illegible]	.045	-.007

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.200	-1.271	-1.419	-1.481	-1.020
	.080	-.988	-.988	-1.092	-1.202	-1.007
	.130	-2.444	-2.648	-2.826	-2.742	-2.760
	.145	-8.687	-8.496	-7.856	-8.563	-8.085
	.155	-3.659	-3.912	-3.872	-3.885	-3.297
	.180	-2.484	-2.509	-2.623	-2.527	-2.538
	.220	-1.589	-1.785	-1.818	-1.891	-1.727
	.270	-1.195	-1.423	-1.446	-1.403	-1.347
	.400	-1.008	-1.100	-1.060	-1.065	-1.217
	.620	-1.088	-1.330	-.661	-1.098	-1.609
	.685	-2.845	-4.946	-.209	-3.073	-9.733
	.693	-2.424	-5.163	-1.511	-1.813	-8.772
	.700	-1.509	-3.642	-1.367	-1.260	-6.535
	.720	-.775	-1.442	-.968	-.975	-2.695
	.750	-.674	-.843	-.968	-.968	-1.805
	.800	-.561	-.659	-.824	-.884	-1.446
	.900	-.487	-.784	-.674	-.786	-1.315
	.980	-.454	-.724	-.687	-.890	-.909
Lower	.025	.494	.718	.700	.702	.556
	.120	.841	.836	.759	.721	.569
	.220	.788	.777	.765	.728	.595
	.300	.661	.711	.713	.637	.497
	.620	.755	.797	.759	.695	.203
	.750	.855	.876	.752	.702	.576
	.850	.641	.665	.556	.552	.445
	.950	.394	.362	.222	.195	.196

TABLE 20 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = 18.9°$											
.032	-.151	.761	-.361	.148	Upper	.010	-4.795	-1.045	-3.470	-3.505	-3.148
.053	-.270	.561	-.484	-.020		.080	-1.772	-1.137	-3.221	-3.355	-1.837
.100	-.165	.321	-.559	-.262		.130	-3.174	-1.011	-3.160	-3.232	-3.280
.145	-.138	.214	-.511	-.216		.145	-9.760	-1.954	-8.250	-8.674	-8.884
.189	-.105	.260	-.443	-.269		.155	-4.202	-1.460	-4.465	-4.453	-3.919
.234	-.072	.307	-.150	-.329		.180	-2.773	-1.958	-3.160	-3.082	-3.076
.280	-.119	.307	.089	-.336		.220	-1.726	-1.123	-2.259	-2.285	-2.154
.326	-.158	.327	.041	-.417		.270	-1.297	-1.669	-1.789	-1.766	-1.758
.371	-.336	.434	-.341	-.484		.400	-.994	-1.242	-1.298	-1.316	-1.614
.392	-.400	.485	-.921	-.578		.620	-.981	-1.416	-.968	-1.316	-1.581
.413	-.461	.534	-1.166	.175		.685	-3.010	-4.160	-.585	-2.516	-7.758
.434	-.547	.561	-1.821	.625		.693	-2.839	-4.427	-1.943	-2.666	-6.915
.457	-.514	.550	-1.418	.706		.700	-1.778	-3.165	-1.715	-1.732	-5.190
.480	-.487	.540	-1.139	.659		.720	-.856	-1.322	-1.224	-1.132	-2.246
.502	-.527	.530	-1.064	.632		.750	-.632	-.788	-1.096	-1.098	-1.640
.551	-.428	.525	-1.009	.598		.800	-.560	.574	-.888	-.962	-1.390
.585	-.389	.521	-1.105	.659		.900	-.435	.534	-.847	-.900	-1.238
.592	-.356	.507	-1.262	-1.177		.980	-.435	.347	-.847	-.921	-.929
.613	-.257	.407	-1.023	-.550							
.634	-.244	.274	-.675	-.437	Lower	.025	.718	.835	.820	.818	.665
.655	-.204	.147	-.402	-.403		.120	.889	.848	.767	.764	.580
.675	-.125	.027	-.218	-.235		.220	.830	.801	.807	.764	.626
.696	-.086	.040	-.089	-.040		.300	.757	.761	.713	.716	.593
.774	.007	.187	.068	.121		.620	.777	.815	.767	.696	.231
.852	-.072	.080	-.048	[illegible]		.750	.856	.868	.746	.702	.606
.930	-.007	.107	-.020	.101		.850	.672	.701	.572	.559	.487
						.950	.448	.454	.188	.232	.224
$\alpha = 22.9°$											
.032	-.227	.809	-.497	.040	Upper	.010	-9.229	-1.038	-4.300	-3.966	-3.880
.053	-.347	.630	-.564	-.107		.080	-2.126	-1.806	-4.407	-3.999	-2.511
.100	-.200	.391	-.603	-.374		.130	-3.474	-1.017	-3.506	-3.263	-2.564
.145	-.160	.298	-.584	-.427		.145	-10.042	-1.196	-6.584	-6.545	-6.737
.189	-.080	.325	-.511	-.347		.155	-4.259	-1.410	-3.980	-3.680	-2.925
.234	-.073	.351	-.166	-.427		.180	-2.817	-1.077	-2.985	-2.672	-2.344
.280	-.120	.378	.133	-.467		.220	-1.747	-1.235	-2.177	-1.969	-1.516
.326	-.220	.385	.046	-.601		.270	-1.469	-.757	-1.729	-1.578	-1.155
.371	-.421	.471	-.444	-.648		.400	-1.090	-.320	-1.362	-1.220	-1.108
.392	-.510	.520	-1.001	-.881		.620	-1.009	-.386	-1.202	-1.061	-1.068
.413	-.601	.584	-1.286	.174		.685	-3.934	-1.614	-.427	-1.936	-2.003
.434	-.628	.617	-2.036	.661		.693	-3.941	-1.972	-1.796	-2.076	-1.576
.457	-.561	.600	-1.552	.714		.700	-2.600	-1.885	-1.589	-1.439	-1.389
.480	-.494	.585	-1.306	.694		.720	-1.293	-.267	-1.068	-1.041	-1.015
.502	-.487	.570	-1.160	.661		.750	-.846	.789	-1.002	-1.021	-1.035
.551	-.327	.555	-1.061	.628		.800	-.609	.524	-.995	-.968	-1.042
.585	-.280	.537	-1.107	.668		.900	-.481	.371	-.895	-.908	-.988
.592	-.267	.524	-1.439	-1.496		.980	-.474	.239	-.835	-.902	-.908
.613	-.174	.411	-1.253	-.650							
.634	-.160	.285	-.796	-.507	Lower	.025	.792	.855	.835	.842	.694
.655	-.134	.139	-.451	-.387		.120	.914	.855	.775	.776	.608
.675	-.080	.027	-.252	-.147		.220	.880	.836	.815	.776	.674
.696	-.020	.073	-.113	-.013		.300	.792	.776	.735	.729	.588
.774	-.027	.095	.027	.107		.620	.799	.836	.761	.710	.280
.852	-.107	.113	-.073	.117		.750	.887	.869	.788	.716	.594
.930	-.027	.139	-.040	.127		.850	.691	.716	.614	.590	.481
						.950	.474	.471	.254	.239	.174

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TABLE 20 Continued (d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface	Surface	x/c	0.221	0.426	0.640	0.800	0.918
	Fuselage						Wing, flap, or aileron				
	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
$\alpha = -1.7°$											
.032	.276	.325	.281	.290	Upper	.010	.915	.851	.883	.818	.795
.053	.051	.104	.052	.046		.080	.369	.221	.158	.150	.256
.100	-.090	-.052	-.098	-.086		.130	-.507	-.825	-.942	-.896	-.808
.145	-.083	-.065	-.065	-.033		.145	-4.795	-4.821	-4.630	-5.226	-4.494
.189	-.013	-.013	-.026	.000		.155	-1.805	-2.001	-2.015	-1.982	-1.481
.234	-.051	.052	.007	.020		.180	-1.383	-1.312	-1.455	-1.361	-1.333
.280	-.064	.078	-.026	.033		.220	-.863	-1.014	-1.093	-1.184	-.910
.326	-.038	.104	-.033	.086		.270	-.698	-.910	-.981	-.916	-.737
.371	-.122	.143	-.150	.145		.400	-.751	-.936	-1.060	-1.027	-.801
.392	-.013	.165	-.020	.375		.620	-1.311	-1.592	-1.581	-1.943	-1.244
.413	-.173	.195	-.209	.099		.685	-6.737	-6.945	-4.195	-7.764	-6.205
.434	-.224	.221	-.451	-.040		.693	-6.658	-7.627	-6.204	-8.719	-5.231
.457	-.263	.240	-.576	-.026		.700	-4.241	-5.438	-4.959	-5.331	-3.918
.480	-.333	.260	-.536	.086		.720	-2.068	-2.404	-2.345	-2.492	-1.641
.502	-.417	.280	-.569	.171		.750	-1.390	-1.475	-1.416	-1.603	-1.269
.551	-.462	.305	-.791	.296		.800	-.994	-.832	-.784	-.903	-1.083
.585	-.449	.331	-.955	.250		.900	-.606	-.234	-.145	-.406	-.885
.592	-.423	.331	-1.119	-1.153		.980	-.053	.260	.079	-.026	-.679
.613	-.308	.247	-.935	-.705	Lower	.025	-.053	.045	.165	.203	.026
.634	-.263	.188	-.680	-.863		.120	-.138	.097	.132	.150	.000
.655	-.218	.130	-.477	-.184		.220	-.099	.071	.092	.124	.000
.675	-.128	.039	-.321	-.059		.300	.059	.052	.040	.072	-.038
.696	-.058	.039	-.216	-.040		.620	.474	.364	.244	.150	-.096
.774	-.045	.078	-.065	.013		.750	.672	.520	.296	.255	.199
.852	-.006	-.026	.007	-.138		.850	.764	.650	.487	.386	.295
.930	.077	-.188	.098	-.270		.950	.580	.598	.474	.392	.186
$\alpha = 5.5°$											
.032	.078	.455	.156	.333	Upper	.010	.491	.455	.404	.409	.458
.053	-.131	.247	-.052	.090		.080	-.265	-.370	-.462	-.513	-.379
.100	-.209	.071	-.195	-.090		.130	-1.492	-1.741	-1.936	-1.930	-1.923
.145	-.157	-.006	-.156	-.071		.145	-7.036	-6.848	-6.551	-7.445	-6.940
.189	-.078	.058	-.097	-.032		.155	-2.905	-3.041	-3.141	-3.216	-2.734
.234	-.105	.110	-.013	-.026		.180	-2.122	-1.975	-2.205	-2.150	-2.237
.280	-.118	.130	.052	-.038		.220	-1.373	-1.462	-1.628	-1.754	-1.596
.326	-.118	.143	.026	-.026		.270	-1.041	-1.273	-1.410	-1.358	-1.334
.371	-.235	.221	-.169	.013		.400	-.935	-1.143	-1.327	-1.351	-1.347
.392	-.285	.255	-.292	.160		.620	-1.160	-1.670	-1.712	-2.241	-2.060
.413	-.334	.299	-.578	.269		.685	-3.554	-6.692	-4.096	-9.044	-12.042
.434	-.399	.351	-.884	.295		.693	-3.117	-7.302	-5.904	-9.901	-11.061
.457	-.425	.365	-.825	.397		.700	-1.790	-5.185	-4.667	-6.289	-8.490
.480	-.484	.380	-.741	.474		.720	-.849	-2.261	-2.212	-3.028	-3.957
.502	-.536	.395	-.715	.455		.750	-.743	-1.358	-1.346	-1.936	-2.852
.551	-.523	.320	-.806	.487		.800	-.663	-.767	-.769	-1.195	-2.258
.585	-.484	.435	-.864	.545		.900	-.570	-.247	-.417	-.526	-1.518
.592	-.471	.429	-.832	-.814		.980	-.471	.130	-.244	-.097	-.563
.613	-.347	.325	-.669	-.555	Lower	.025	.192	.429	.333	.487	.131
.634	-.314	.227	-.507	-.513		.120	.285	.422	.397	.455	.157
.655	-.255	.117	-.377	-.468		.220	.557	.533	.667	.526	.484
.675	-.157	-.006	-.240	-.192		.300	.617	.611	.622	.585	.477
.696	-.098	.032	-.136	-.038		.620	.703	.728	.679	.676	.013
.774	-.046	.097	.006	.019		.750	.796	.793	.718	.702	.517
.852	-.059	.052	-.019	-.122		.850	.597	.663	.603	.617	.458
.930	.026	-.110	.052	-.135		.950	.371	.533	.359	.461	.275
$\alpha = 13.0°$											
.032	-.079	.632	-.106	.262	Upper	.010	-.341	-1.627	-2.010	-2.135	-1.791
.053	-.211	.435	-.292	.067		.080	-1.105	-1.120	-1.264	-1.406	-1.238
.100	-.184	.211	-.371	-.168		.130	-2.632	-2.871	-3.254	-3.203	-3.260
.145	-.125	.125	-.292	-.202		.145	-9.070	-8.944	-8.808	-9.682	-9.168
.189	-.079	.165	-.292	-.155		.155	-3.867	-4.175	-4.489	-4.589	-3.925
.234	-.105	.224	-.093	-.182		.180	-2.632	-2.746	-3.113	-3.084	-3.043
.280	-.119	.237	.073	-.208		.220	-1.718	-1.976	-2.253	-2.387	-2.147
.326	-.132	.244	.033	-.269		.270	-1.316	-1.620	-1.869	-1.857	-1.765
.371	-.277	.329	-.225	-.289		.400	-1.112	-1.311	-1.594	-1.651	-1.686
.392	-.325	.380	-.650	-.242		.620	-1.262	-1.791	-1.977	-2.467	-2.489
.413	-.382	.435	-.988	.202		.685	-2.782	-5.861	-3.947	-9.708	-14.239
.434	-.441	.487	-1.326	.551		.693	-2.237	-6.178	-5.628	-9.861	-13.251
.457	-.454	.487	-1.121	.632		.700	-1.405	-4.360	-4.431	-6.286	-10.281
.480	-.481	.487	-.968	.592		.720	-.641	-1.851	-2.091	-3.031	-4.834
.502	-.533	.487	-.902	.551		.750	-.573	-1.021	-1.284	-1.910	-3.477
.551	-.487	.487	-.935	.592		.800	-.477	-.514	-.787	-1.154	-2.674
.585	-.441	.487	-.942	.652		.900	-.409	-.198	-.592	-.590	-1.712
.592	-.395	.481	-.942	-.995		.980	-.389	-.072	-.430	-.517	-.586
.613	-.283	.375	-.736	-.450	Lower	.025	.511	.731	.733	.756	.560
.634	-.263	.257	-.504	-.390		.120	.832	.823	.773	.716	.514
.655	-.224	.132	-.345	-.417		.220	.784	.777	.773	.716	.580
.675	-.145	.000	-.199	-.276		.300	.696	.711	.693	.670	.487
.696	-.079	.033	-.119	-.108		.620	.771	.790	.760	.676	-.013
.774	-.013	.165	.033	.067		.750	.859	.883	.746	.676	.501
.852	-.059	.086	-.027	-.054		.850	.641	.738	.659	.630	.421
.930	.020	.040	.013	.007		.950	.396	.540	.377	.418	.257

TABLE 20 Concluded
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = 18.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.171	.769	-.336	.125
.053	-.290	.577	-.491	-.048
.100	-.191	.343	-.551	-.311
.145	-.151	.220	-.524	-.325
.189	-.092	.261	-.464	-.284
.234	-.079	.302	-.161	-.339
.280	-.125	.323	.094	-.380
.326	-.165	.343	.067	-.463
.371	-.356	.426	-.323	-.519
.392	-.415	.475	-.908	-.657
.413	-.487	.563	-1.157	.180
.434	-.540	.604	-1.856	.643
.457	-.527	.595	-1.412	.726
.480	-.514	.585	-1.163	.692
.502	-.527	.575	-1.083	.664
.551	-.454	.565	-1.009	.629
.585	-.421	.543	-1.123	.685
.592	-.375	.529	-1.224	-1.224
.613	-.270	.398	-.908	-.400
.634	-.237	.302	-.605	-.374
.655	-.191	.172	-.363	-.380
.675	-.119	.041	-.208	-.235
.696	-.059	.055	-.101	-.083
.774	.000	.199	.040	-.020
.852	-.079	.103	-.061	.014
.930	-.026	.124	-.027	.104

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-5.381	-3.455	-4.068	-4.061	-3.688
	.080	-1.909	-2.637	-3.957	-4.055	-2.463
	.130	-3.348	-3.173	-3.452	-3.597	-3.635
	.145	-10.175	-9.382	-8.889	-8.889	-9.826
	.155	-4.385	-4.732	-4.946	-4.828	-4.505
	.180	-2.966	-3.228	-3.590	-3.490	-3.556
	.220	-1.828	-2.308	-2.663	-2.663	-2.562
	.270	-1.432	-1.861	-2.179	-2.179	-2.121
	.400	-1.057	-1.429	-1.771	-1.822	-2.022
	.620	-1.159	-1.854	-2.103	-2.508	-2.667
	.685	-3.001	-4.897	-3.687	-7.154	-13.876
	.693	-2.769	-5.137	-5.368	-7.410	-12.849
	.700	-1.718	-1.578	-4.227	-4.714	-9.852
	.720	-.798	-1.490	-2.075	-2.367	-4.557
	.750	-.600	-.810	-1.328	-1.513	-3.220
	.800	-.539	-.412	-.796	-.908	-2.450
	.900	-.416	-.220	-.374	-.565	-1.600
	.980	-.375	-.144	-.194	-.524	-.659
Lower	.025	.723	.865	.823	.780	.606
	.120	.900	.859	.782	.686	.481
	.220	.859	.817	.823	.726	.553
	.300	.764	.804	.754	.679	.474
	.620	.825	.845	.789	.652	-.072
	.750	.880	.907	.740	.666	.487
	.850	.702	.762	.685	.592	.421
	.950	.450	.536	.436	.343	.250

$\alpha = 22.9°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.242	.841	-.447	.020
.053	-.360	.632	-.535	-.141
.100	-.196	.397	-.603	-.390
.145	-.157	.303	-.555	-.424
.189	-.065	.336	-.501	-.370
.234	-.092	.370	-.156	-.464
.280	-.137	.363	.135	-.471
.326	-.249	.390	-.014	-.585
.371	-.438	.437	-.467	-.666
.392	-.530	.515	-.989	-.847
.413	-.615	.598	-1.287	.182
.434	-.628	.625	-2.072	.652
.457	-.563	.605	-1.585	.733
.480	-.504	.585	-1.341	.713
.502	-.484	.565	-1.185	.679
.551	-.314	.545	-1.077	.625
.585	-.262	.538	-1.171	.666
.592	-.249	.538	-1.537	-1.499
.613	-.170	.430	-1.314	-.650
.634	-.170	.289	-.799	-.511
.655	-.144	.155	-.427	-.390
.675	-.085	.020	-.237	-.128
.696	-.046	.054	-.135	.034
.774	-.052	.075	.014	.134
.852	-.144	.101	-.074	.138
.930	-.039	.148	-.020	.141

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-9.261	-4.129	-4.391	-4.117	-3.800
	.080	-2.128	-3.907	-4.512	-4.117	-2.551
	.130	-3.451	-3.053	-3.530	-3.460	-2.479
	.145	-10.086	-8.230	-7.107	-6.243	-6.639
	.155	-4.269	-4.465	-4.243	-3.636	-2.878
	.180	-2.803	-3.113	-3.187	-2.688	-2.309
	.220	-1.739	-2.266	-2.333	-2.018	-1.504
	.270	-1.453	-1.789	-1.883	-1.639	-1.158
	.400	-1.098	-1.338	-1.466	-1.320	-1.105
	.620	-1.043	-1.365	-1.345	-1.050	-1.053
	.685	-3.853	-3.503	-.793	-2.031	-2.002
	.693	-3.894	-4.014	-2.098	-2.160	-1.681
	.700	-2.571	-2.891	-1.802	-1.530	-1.491
	.720	-1.268	-1.257	-1.130	-1.124	-1.060
	.750	-.846	-.760	-1.036	-1.117	-1.079
	.800	-.607	-.518	-.982	-1.070	-1.073
	.900	-.471	-.390	-.901	-.962	-1.001
	.980	-.484	-.309	-.807	-.962	-.929
Lower	.025	.818	.861	.841	.819	.667
	.120	.921	.861	.773	.758	.589
	.220	.887	.827	.827	.792	.648
	.300	.812	.793	.753	.772	.569
	.620	.818	.854	.780	.738	.255
	.750	.887	.874	.760	.718	.549
	.850	.702	.733	.619	.596	.445
	.950	.491	.504	.249	.196	.150

L-927

L-927

TABLE 21 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = -1.5°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.251	.300	.298	.266
.053	.043	.092	.074	.054
.100	-.092	-.043	-.093	-.091
.145	-.080	-.067	-.081	-.036
.189	-.018	-.018	-.025	-.012
.234	-.055	.055	-.006	.012
.280	-.067	.067	-.037	.030
.326	-.043	.080	-.025	.073
.371	-.135	.116	-.013	.133
.392	-.024	.135	.000	.333
.413	-.153	.147	-.167	-.103
.434	-.208	.153	-.409	-.236
.457	-.239	.180	-.521	-.151
.480	-.294	.210	-.471	-.006
.502	-.380	.240	-.496	.115
.551	-.410	.265	-.676	.296
.585	-.404	.288	-.806	.260
.592	-.386	.300	-.931	-.919
.613	-.282	.227	-.769	-.677
.634	-.251	.178	-.527	-.750
.655	-.202	.098	-.360	-.163
.675	-.122	.024	-.230	-.054
.696	-.055	-.061	-.149	-.030
.774	-.031	.012	-.050	.030
.852	.000	-.043	.012	-.103
.930	.067	-.184	.087	-.212

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.962	.876	.841	.831	.827
	.080	.465	.343	.314	.248	.325
	.130	-.383	-.619	-.677	-.695	-.631
	.145	-4.493	-4.226	-3.913	-4.516	-3.944
	.155	-1.672	-1.678	-1.627	-1.638	-1.219
	.180	-1.263	-1.084	-1.149	-1.098	-1.115
	.220	-.754	-.790	-.804	-.918	-.729
	.270	-.616	-.704	-.695	-.658	-.570
	.400	-.641	-.698	-.635	-.639	-.570
	.620	-1.156	-1.145	-.314	-.794	-.919
	.685	-6.366	-5.236	-.115	-2.457	-5.922
	.693	-6.285	-5.690	-1.397	-2.550	-4.912
	.700	-3.984	-4.054	-1.228	-1.675	-3.473
	.720	-1.904	-1.641	-.913	-.918	-1.194
	.750	-1.263	-.949	-.913	-.931	-.851
	.800	-.842	-.643	-.695	-.887	-.759
	.900	-.559	-.551	-.635	-.788	-.692
	.980	-.025	-.484	-.635	-.757	-.588
Lower	.025	-.220	-.141	.054	.099	-.031
	.120	-.308	-.104	.042	.050	-.031
	.220	-.239	-.116	.006	.043	-.024
	.300	-.075	-.171	-.054	-.025	-.073
	.620	.471	.410	.103	.025	-.184
	.750	.754	.631	.151	.099	.116
	.850	.742	.692	.321	.236	.263
	.950	.547	.423	.200	.217	.257

$\alpha = 5.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.087	.509	.124	.312
.053	-.112	.274	-.085	.097
.100	-.194	.096	-.209	-.117
.145	-.156	.013	-.170	-.078
.189	-.069	.057	-.111	-.026
.234	-.106	.134	-.020	-.039
.280	-.106	.140	.039	-.045
.326	-.112	.166	.026	-.032
.371	-.187	.217	-.164	.006
.392	-.240	.255	-.268	.156
.413	-.287	.293	-.569	.156
.434	-.362	.344	-.877	.208
.457	-.400	.360	-.831	.357
.480	-.437	.380	-.746	.442
.502	-.512	.400	-.720	.442
.551	-.512	.420	-.824	.513
.585	-.481	.433	-.962	.533
.592	-.443	.439	-1.020	-1.033
.613	-.306	.357	-.778	-.617
.634	-.300	.267	-.530	-.624
.655	-.244	.166	-.347	-.078
.675	-.144	.038	-.222	-.013
.696	-.106	.025	-.131	.000
.774	-.012	.045	-.020	.045
.852	-.025	.070	.026	-.097
.930	.050	-.115	.085	-.156

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	.603	.586	.539	.595	.599
	.080	-.170	-.229	-.253	-.301	-.200
	.130	-1.307	-1.554	-1.650	-1.622	-1.548
	.145	-6.605	-6.355	-5.990	-6.718	-5.925
	.155	-2.709	-2.770	-2.755	-2.747	-2.217
	.180	-1.967	-1.808	-1.897	-1.812	-1.829
	.220	-1.232	-1.286	-1.332	-1.459	-1.255
	.270	-.949	-1.089	-1.117	-1.066	-.987
	.400	-.855	-.930	-.916	-.922	-.930
	.620	-1.232	-1.274	-.500	-1.027	-1.324
	.685	-5.562	-5.384	-.286	-2.270	-8.810
	.693	-5.336	-5.756	-1.702	-2.453	-7.942
	.700	-3.331	-4.088	-1.488	-1.584	-5.957
	.720	-1.659	-1.668	-1.027	-.988	-2.518
	.750	-1.219	-.981	-1.040	-1.053	-1.723
	.800	-.930	-.700	-.695	-.975	-1.386
	.900	-.478	-.700	-.582	-.831	-1.136
	.980	-.151	-.624	-.656	-.877	-.599
Lower	.025	.126	.428	.344	.399	.194
	.120	.189	.331	.292	.242	.100
	.220	.503	.446	.338	.301	.156
	.300	.591	.560	.533	.497	.468
	.620	.685	.713	.669	.680	.287
	.750	.792	.796	.702	.726	.618
	.850	.628	.630	.552	.556	.506
	.950	.452	.325	.188	.183	.262

$\alpha = 13.2°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.091	.641	-.124	.255
.053	-.247	.436	-.294	.045
.100	-.208	.218	-.379	-.185
.145	-.143	.128	-.347	-.172
.189	-.104	.154	-.275	-.134
.234	-.110	.218	-.078	-.166
.280	-.123	.218	.059	-.185
.326	-.156	.244	.033	-.223
.371	-.273	.321	-.235	-.223
.392	-.330	.380	-.628	-.191
.413	-.396	.436	-.968	.197
.434	-.455	.468	-1.269	.516
.457	-.448	.470	-1.073	.605
.480	-.481	.470	-.929	.560
.502	-.533	.472	-.837	.516
.551	-.487	.474	-.883	.554
.585	-.448	.474	-.877	.618
.592	-.429	.474	-.890	-.847
.613	-.286	.378	-.720	-.520
.634	-.286	.244	-.517	-.433
.655	-.260	.122	-.347	-.414
.675	-.169	.019	-.203	-.204
.696	-.117	.019	-.098	-.038
.774	-.006	.077	.033	.083
.852	-.078	.077	-.033	-.057
.930	.013	.013	.020	-.006

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.201	-1.224	-1.433	-1.511	-.773
	.080	-.988	-.981	-1.076	-1.217	-.988
	.130	-2.443	-2.577	-2.802	-2.819	-2.742
	.145	-8.628	-8.269	-7.813	-8.791	-8.011
	.155	-3.625	-3.788	-3.840	-3.957	-3.266
	.180	-2.475	-2.442	-2.624	-2.610	-2.553
	.220	-1.559	-1.731	-1.821	-1.962	-1.715
	.270	-1.202	-1.397	-1.458	-1.465	-1.345
	.400	-.988	-1.077	-1.057	-1.138	-1.221
	.620	-1.085	-1.295	-.675	-1.138	-1.624
	.685	-2.898	-4.808	-.331	-2.453	-9.784
	.693	-2.391	-5.032	-1.573	-2.159	-8.862
	.700	-1.481	-3.558	-1.356	-1.518	-6.607
	.720	-.767	-1.410	-.962	-1.092	-2.742
	.750	-.663	-.795	-.891	-1.040	-1.852
	.800	-.546	-.590	-.796	-.935	-1.488
	.900	-.494	-.699	-.650	-.805	-1.325
	.980	-.455	-.679	-.656	-.863	-.916
Lower	.025	.468	.692	.681	.680	.546
	.120	.806	.808	.751	.706	.578
	.220	.767	.737	.732	.720	.598
	.300	.656	.667	.656	.628	.481
	.620	.747	.763	.745	.680	.182
	.750	.845	.859	.720	.720	.578
	.850	.604	.654	.541	.556	.461
	.950	.390	.353	.223	.203	.195

TABLE 21 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 2.0 h_d/c = 1.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.[illegible]26	0.640	0.800	0.918
α = 18.8°											
.032	-.206	.754	-.308	.164	Upper	.010	-4.468	-2.995	-3.207	-3.340	-3.269
.053	-.325	.546	-.442	-.032		.080	-1.786	-2.105	-2.897	-3.231	-1.857
.100	-.212	.325	-.513	-.259		.130	-3.199	-2.963	-3.017	-3.103	-3.375
.145	-.179	.234	-.462	-.304		.145	-9.851	-8.855	-7.996	-8.154	-9.065
.189	-.113	.260	-.429	-.247		.155	-4.232	-4.405	-4.276	-4.199	-4.052
.234	-.106	.292	-.103	-.310		.180	-2.813	-2.924	-3.024	-2.923	-3.223
.280	-.153	.305	.083	-.329		.220	-1.740	-2.098	-2.157	-2.167	-2.281
.326	-.186	.325	.064	-.411		.270	-1.321	-1.663	-1.702	-1.679	-1.850
.371	-.351	.416	-.301	-.455		.400	-1.001	-1.234	-1.240	-1.256	-1.638
.392	-.425	.475	-.846	-.563		.620	-1.007	-1.[illegible]16	-.886	-1.128	-1.651
.413	-.504	.526	-1.115	.158		.685	-3.146	-4.165	-.620	-1.846	-7.235
.434	-.557	.565	-1.724	.595		.693	-2.957	-4.[illegible]22	-1.891	-2.058	-6.393
.457	-.544	.560	-1.372	.671		.700	-1.845	-3.[illegible]96	-1.638	-1.462	-4.874
.480	-.537	.545	-1.077	.645		.720	-.922	-1.[illegible]51	-1.177	-1.071	-2.195
.502	-.557	.530	-1.000	.601		.750	-.700	-.[illegible]93	-1.025	-1.019	-1.631
.551	-.438	.515	-.949	.563		.800	-.589	-.[illegible]46	-.886	-.846	-1.419
.585	-.405	.507	-1.019	.620		.900	-.504	-.[illegible]61	-.797	-.763	-1.280
.592	-.378	.500	-1.154	-1.132		.980	-.425	-.[illegible]60	-.778	-.744	-.975
.613	-.232	.396	-.917	-.700	Lower	.025	.680	.[illegible]32	.765	.782	.643
.634	-.245	.273	-.622	-.405		.120	.883	.419	.721	.718	.570
.655	-.219	.130	-.372	-.392		.220	.831	.[illegible]80	.759	.737	.603
.675	-.126	.013	-.218	-.228		.300	.733	.[illegible]54	.696	.679	.544
.696	-.086	.026	-.090	-.057		.620	.765	.606	.734	.673	.186
.774	-.007	.110	.038	.101		.750	.850	.838	.715	.679	.557
.852	-.093	.104	-.045	-.013		.850	.641	.689	.569	.577	.444
.930	-.046	.104	-.032	.089		.950	.406	.442	.228	.244	.179
α = 22.9°											
.032	-.252	.815	-.430	.007	Upper	.010	-8.809	-4.[illegible]80	-4.208	-3.808	-3.786
.053	-.365	.628	-.525	-.119		.080	-2.059	-3.[illegible]60	-4.314	-3.852	-2.480
.100	-.212	.387	-.607	-.382		.130	-3.311	-3.[illegible]72	-3.332	-3.125	-2.606
.145	-.172	.294	-.557	-.415		.145	-9.699	-8.[illegible]07	-6.981	-6.218	-6.744
.189	-.086	.307	-.487	-.369		.155	-4.098	-4.[illegible]87	-4.123	-3.479	-2.951
.234	-.066	.367	-.120	-.428		.180	-2.723	-3.[illegible]12	-3.043	-2.537	-2.374
.280	-.126	.354	.139	-.468		.220	-1.697	-2.[illegible]50	-2.213	-1.866	-1.532
.326	-.212	.387	.070	-.573		.270	-1.407	-1.[illegible]96	-1.758	-1.474	-1.154
.371	-.431	.481	-.411	-.652		.400	-1.045	-1.[illegible]49	-1.311	-1.145	-1.107
.392	-.510	.530	-.955	-.876		.620	-.981	-1.[illegible]89	-1.166	-.911	-1.068
.413	-.610	.588	-1.240	.198		.685	-3.730	-3.519	-.698	-1.904	-1.963
.434	-.663	.628	-1.980	.665		.693	-3.665	-3.873	-2.002	-2.031	-1.631
.457	-.584	.600	-1.518	.744		.700	-2.420	-2.831	-1.765	-1.385	-1.412
.480	-.511	.580	-1.284	.705		.720	-1.226	-1.242	-1.146	-.968	-1.015
.502	-.511	.560	-1.139	.672		.750	-.839	-.801	-1.041	-.962	-1.068
.551	-.345	.540	-1.031	.626		.800	-.587	-.[illegible]58	-.948	-.911	-1.028
.585	-.292	.534	-1.139	.659		.900	-.439	-.451	-.803	-.854	-.988
.592	-.279	.514	-1.455	-1.502		.980	-.400	-.[illegible]57	-.738	-.854	-.915
.613	-.220	.414	-1.215	-.600	Lower	.025	.794	.858	.843	.816	.683
.634	-.179	.280	-.746	-.487		.120	.891	.858	.790	.740	.603
.655	-.166	.147	-.392	-.395		.220	.852	.815	.830	.778	.676
.675	-.080	.013	-.221	-.145		.300	.774	.7[illegible]8	.738	.715	.603
.696	-.046	.053	-.101	[illegible]		.620	.794	.8[illegible]5	.790	.715	.285
.774	-.027	.134	.025	.125		.750	.858	.8[illegible]1	.797	.721	.610
.852	-.139	.100	-.070	.128		.850	.684	.7[illegible]5	.612	.576	.484
.930	-.046	.120	-.032	.132		.950	.484	.4[illegible]1	.263	.240	.159

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TABLE 21(b) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.5$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

$\alpha = -1.7°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.264	.314	.275	.314
.053	.044	.082	.050	.057
.100	-.113	-.025	-.087	-.082
.145	-.101	-.075	-.062	-.044
.189	-.019	[illegible]	[illegible]	[illegible]
.234	-.069	.044	.019	.013
.280	-.063	.075	-.031	.025
.326	-.057	.094	-.012	.063
.371	-.119	.132	-.130	.138
.392	-.031	.150	.	.352
.413	-.195	.170	-.175	.
.434	-.245	.189	-.443	-.151
.457	-.270	.210	-.524	-.075
.480	-.339	.240	-.500	.075
.502	-.421	.270	-.543	.182
.551	-.459	.300	-.737	.339
.585	-.440	.314	-.899	.308
.592	-.421	.308	-1.049	-1.006
.613	-.302	.251	-.862	-.850
.634	-.270	.182	-.612	-.792
.655	-.220	.132	-.412	-.170
.675	-.138	.050	-.275	-.069
.696	-.069	.031	-.187	-.031
.774	-.075	-.006	-.075	.013
.852	-.044	-.025	-.006	-.126
.930	.063	-.207	.094	-.277

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.918	.848	.823	.824	.817
	.080	.401	.270	.226	.200	.270
	.130	-.419	-.748	-.848	-.818	-.760
	.145	-4.580	-4.607	-4.399	-4.895	-4.305
	.155	-1.732	-1.898	-1.904	-1.861	-1.427
	.180	-1.294	-1.238	-1.376	-1.249	-1.288
	.220	-.820	-.930	-1.018	-1.080	-.880
	.270	-.647	-.836	-.905	-.818	-.698
	.400	-.703	-.867	-.955	-.893	-.742
	.620	-1.251	-1.489	-1.288	-1.536	-1.163
	.685	-6.540	-6.649	-3.387	-6.575	-5.933
	.693	-6.472	-7.234	-4.764	-7.255	-4.933
	.700	-4.130	-5.191	-3.626	-4.333	-3.626
	.720	-2.046	-2.294	-1.640	-1.873	-1.489
	.750	-1.381	-1.395	-1.018	-1.143	-1.144
	.800	-.937	-.798	-.710	-.649	-.987
	.900	-.573	-.245	-.434	-.412	-.823
	.980	-.012	.189	-.277	-.125	-.647
Lower	.025	-.142		.132	.187	.019
	.120	-.216	.025	.119	.087	-.019
	.220	-.154	-.025	.069	.081	-.025
	.300	.031	-.044	[illegible]	.044	-.069
	.620	.487	.408	.163	.100	-.057
	.750	.764	.584	.283	.200	.182
	.850	.721	.654	.446	.343	.270
	.950	.549	.584	.377	.362	.170

$\alpha = 5.5°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.084	.509	.140	.336
.053	-.117	.261	-.076	.103
.100	-.195	.070	-.217	-.090
.145	-.143	.013	-.166	-.058
.189	-.078	.064	-.096	-.026
.234	-.110	.140	-.019	-.039
.280	-.110	.146	.045	-.026
.326	-.110	.153	.006	-.019
.371	-.201	.223	-.166	.013
.392	-.265	.255	-.287	.168
.413	-.318	.299	-.567	.232
.434	-.383	.344	-.872	.258
.457	-.403	.360	-.821	.413
.480	-.455	.380	-.745	.458
.502	-.539	.400	-.720	.478
.551	-.520	.415	-.802	.497
.585	-.481	.439	-.872	.555
.592	-.461	.427	-.828	-.839
.613	-.318	.337	-.669	-.700
.634	-.299	.229	-.509	-.555
.655	-.260	.134	-.369	-.465
.675	-.162	.019	-.261	-.168
.696	-.123	.032	-.153	-.019
.774	-.026	.038	.006	.019
.852	-.058	.032	-.038	-.123
.930	.039	-.089	.045	-.129

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	.567	.560	.452	.452	.520
	.080	-.229	-.261	-.374	-.439	-.286
	.130	-1.388	-1.598	-1.859	-1.828	-1.767
	.145	-6.794	-6.559	-6.440	-7.164	-6.542
	.155	-2.808	-2.885	-3.027	-3.031	-2.547
	.180	-2.050	-1.879	-2.142	-2.031	-2.118
	.220	-1.286	-1.375	-1.549	-1.637	-1.462
	.270	-.993	-1.178	-1.329	-1.261	-1.208
	.400	-.879	-1.063	-1.226	-1.210	-1.195
	.620	-1.133	-1.547	-1.420	-1.764	-1.754
	.685	-3.744	-6.387	-3.207	-6.782	-10.655
	.693	-3.318	-6.858	-4.401	-7.418	-9.745
	.700	-1.923	-4.916	-3.240	-4.362	-7.419
	.720	-.898	-2.133	-1.504	-1.847	-3.352
	.750	-.751	-1.254	-.955	-1.280	-2.404
	.800	-.662	-.694	-.774	-.872	-1.923
	.900	-.573	-.248	-.697	-.783	-1.442
	.980	-.465	.083	-.555	-.650	-.624
Lower	.025	.153	.420	.374	.433	.195
	.120	.217	.388	.348	.325	.084
	.220	.529	.478	.523	.433	.292
	.300	.599	.599	.645	.573	.494
	.620	.700	.732	.697	.656	.182
	.750	.815	.771	.723	.713	.533
	.850	.592	.669	.594	.579	.481
	.950	.369	.490	.277	.318	.266

$\alpha = 13.1°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.078	.663	-.078	.263
.053	-.240	.444	-.275	.046
.100	-.195	.206	-.386	-.198
.145	-.136	.113	-.334	-.178
.189	-.084	.146	-.275	-.138
.234	-.110	.199	-.072	-.171
.280	-.136	.219	.078	-.204
.326	-.143	.239	.059	-.237
.371	-.279	.325	-.216	-.237
.392	.084	.385	-.628	-.217
.413	-.403	.431	-.988	.211
.434	-.468	.484	-1.315	.540
.457	-.468	.486	-1.105	.626
.480	-.481	.488	-.962	.580
.502	-.559	.490	-.883	.547
.551	-.494	.491	-.916	.580
.585	-.455	.491	-.877	.593
.592	-.416	.477	-.863	-.843
.613	-.286	.371	-.674	-.500
.634	-.299	.265	-.491	-.389
.655	-.240	.139	-.327	-.421
.675	-.156	-.013	-.203	-.277
.696	-.110	.013	-.098	-.092
.774	-.013	.066	.059	.072
.852	-.078	.060	-.026	-.059
.930	.013	.020	.	.

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.257	-1.459	-1.765	-1.799	-1.533
	.080	-1.034	-1.088	-1.238	-1.295	-1.156
	.130	-2.529	-2.812	-3.135	-3.015	-3.054
	.145	-8.779	-8.879	-8.568	-9.269	-8.660
	.155	-3.721	-4.111	-4.307	-4.291	-3.625
	.180	-2.555	-2.692	-2.983	-2.845	-2.846
	.220	-1.660	-1.936	-2.154	-2.172	-1.962
	.270	-1.271	-1.572	-1.758	-1.668	-1.585
	.400	-1.067	-1.273	-1.449	-1.439	-1.481
	.620	-1.179	-1.711	-1.620	-1.851	-2.040
	.685	-2.839	-6.107	-3.247	-6.463	-11.305
	.693	-2.226	-6.260	-4.195	-6.116	-10.317
	.700	-1.403	-4.443	-3.082	-3.362	-7.822
	.720	-.672	-1.870	-1.455	-1.432	-3.495
	.750	-.599	-1.034	-.962	-1.197	-2.495
	.800	-.501	-.544	-.764	-.896	-2.027
	.900	-.428	-.206	-.751	-.844	-1.715
	.980	-.520	-.199	-.751	-1.001	-1.189
Lower	.025	.487	.736	.718	.739	.578
	.120	.790	.822	.744	.726	.559
	.220	.757	.769	.757	.726	.591
	.300	.672	.703	.678	.648	.494
	.620	.744	.822	.744	.687	.078
	.750	.843	.908	.751	.726	.526
	.850	.626	.723	.619	.589	.429
	.950	.382	.531	.277	.268	.143

TABLE 21 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 1.0$ $h_d/c = 0.5$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface	Surface	x/c	0.221	0.426	0.640	0.800	0.918
	Fuselage						Wing, flap, or aileron				
$\alpha = 18.7°$											
.032	-.177	.724	-.301	.153	Upper	.010	-4.691	-3.141	-3.627	-3.709	-3.545
.053	-.327	.527	-.438	-.033	Upper	.080	-1.780	-2.239	-3.402	-3.630	-2.276
.100	-.222	.323	-.517	-.292	Upper	.130	-3.170	-3.030	-3.263	-3.310	-3.499
.145	-.170	.211	-.484	-.298	Upper	.145	-9.713	-9.069	-8.488	-8.798	-9.354
.189	-.105	.237	-.419	-.265	Upper	.155	-4.197	-4.518	-4.668	-4.618	-4.304
.234	-.092	.277	-.118	-.338	Upper	.180	-2.820	-3.023	-3.329	-3.264	-3.414
.280	-.137	.303	.118	-.345	Upper	.220	-1.748	-2.200	-2.447	-2.473	-2.486
.326	-.177	.310	.085	-.451	Upper	.270	-1.332	-1.752	-1.976	-1.975	-2.041
.371	-.353	.421	-.294	-.517	Upper	.400	-1.001	-1.330	-1.552	-1.603	-1.832
.392	-.425	.480	-.890	-.610	Upper	.620	-1.040	-1.633	-1.645	-1.812	-1.975
.413	-.504	.540	-1.132	.166	Upper	.685	-2.878	-4.742	-2.931	-3.905	-9.432
.434	-.563	.566	-1.773	.630	Upper	.693	-2.638	-5.[illegible]45	-4.105	-4.023	-8.477
.457	-.530	.555	-1.393	.696	Upper	.700	-1.644	-3.[illegible]76	-3.143	-2.335	-6.410
.480	-.517	.543	-1.105	.676	Upper	.720	-.767	-1.[illegible]21	-1.532	-1.262	-2.891
.502	-.549	.530	-1.020	.637	Upper	.750	-.585	-.[illegible]23	-1.048	-1.138	-2.100
.551	-.445	.515	-.955	.603	Upper	.800	-.520	-.[illegible]28	-.796	-.909	-1.792
.585	-.406	.507	-1.079	.663	Upper	.900	-.396	-.[illegible]11	-.743	-.805	-1.563
.592	-.379	.507	-1.151	-1.147	Upper	.980	-.403	-.[illegible]71	-.696	-.824	-1.171
.613	-.249	.408	-.870	-.500	Lower	.025	.682	.[illegible]17	.816	.785	.602
.634	-.242	.283	-.595	-.378	Lower	.120	.858	.[illegible]43	.749	.720	.517
.655	-.196	.165	-.366	-.378	Lower	.220	.812	.[illegible]90	.796	.739	.595
.675	-.118	.026	-.196	-.245	Lower	.300	.728	.[illegible]51	.743	.680	.523
.696	-.072	.040	-.098	-.073	Lower	.620	.767	.[illegible]30	.769	.680	.111
.774	-.013	.105	.052	.133	Lower	.750	.832	.869	.763	.706	.517
.852	-.092	.086	-.052	.129	Lower	.850	.637	.731	.637	.602	.445
.930	-.039	.105	-.007	.126	Lower	.950	.422	.514	.312	.281	.150
$\alpha = 22.9°$											
.032	-.252	.752	-.451	.040	Upper	.010	-8.858	-3.[illegible]51	-4.248	-4.075	-3.747
.053	-.351	.536	-.565	-.112	Upper	.080	-2.088	-3.[illegible]89	-4.360	-4.075	-2.513
.100	-.219	.386	-.612	-.362	Upper	.130	-3.418	-3.[illegible]02	-3.398	-3.423	-2.527
.145	-.199	.288	-.572	-.408	Upper	.145	-9.997	-8.[illegible]65	-6.744	-6.408	-6.678
.189	-.099	.301	-.518	-.356	Upper	.155	-4.248	-4.[illegible]50	-4.044	-3.651	-2.911
.234	-.073	.321	-.155	-.421	Upper	.180	-2.799	-3.[illegible]35	-2.997	-2.710	-2.314
.280	-.133	.360	.114	-.454	Upper	.220	-1.726	-2.[illegible]11	-2.200	-2.024	-1.505
.326	-.219	.373	-.034	-.553	Upper	.270	-1.436	-1.[illegible]40	-1.758	-1.620	-1.147
.371	-.431	.484	-.477	-.652	Upper	.400	-1.073	-1.[illegible]15	-1.350	-1.304	-1.107
.392	-.520	.530	-.988	-.856	Upper	.620	-1.014	-1.[illegible]28	-1.179	-1.002	-1.034
.413	-.617	.589	-1.298	.178	Upper	.685	-3.787	-3.[illegible]01	-.665	-2.091	-2.122
.434	-.663	.634	-2.078	.672	Upper	.693	-3.800	-3.[illegible]79	-1.923	-2.226	-1.777
.457	-.577	.610	-1.587	.738	Upper	.700	-2.509	-2.[illegible]00	-1.666	-1.580	-1.565
.480	-.511	.590	-1.318	.692	Upper	.720	-1.264	-1.[illegible]17	-1.073	-1.116	-1.094
.502	-.511	.570	-1.170	.659	Upper	.750	-.843	-.[illegible]52	-1.008	-1.109	-1.081
.551	-.318	.550	-1.076	.632	Upper	.800	-.599	-.[illegible]04	-.955	-1.069	-1.081
.585	-.279	.536	-1.136	.652	Upper	.900	-.454	-.[illegible]53	-.863	-.988	-1.008
.592	-.245	.491	-1.506	-1.403	Upper	.980	-.474	-.[illegible]68	-.771	-.955	-.948
.613	-.126	.399	-1.304	-.700	Lower	.025	.797	.[illegible]70	.843	.827	.656
.634	-.172	.307	-.760	-.494	Lower	.120	.909	.[illegible]70	.784	.787	.577
.655	-.139	.150	-.437	-.362	Lower	.220	.869	.[illegible]18	.823	.787	.663
.675	-.073	.033	-.235	-.132	Lower	.300	.797	.[illegible]05	.751	.773	.577
.696	-.046	.052	-.128	.020	Lower	.620	.810	.[illegible]31	.777	.713	.272
.774	-.046	.144	.027	.132	Lower	.750	.869	.[illegible]70	.777	.713	.577
.852	-.146	.098	-.074	.136	Lower	.850	.705	.[illegible]39	.612	.558	.471
.930	-.060	.137	-.034	.138	Lower	.950	.468	.[illegible]84	.270	.182	.153

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TABLE 21 (c) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 2.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	Fuselage, 0.000, Upper surface	Fuselage, 0.000, Lower surface	Fuselage, 0.154, Upper surface	Fuselage, 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron, 0.221	0.426	0.640	0.800	0.918
							C_p values for spanwise stations, $\frac{y}{b/2}$, of:				
$\alpha = -1.5°$											
.032	.256	.321	.285	.292	Upper	.010	.962	.871	.840	.844	.830
.053	.031	.091	.081	.073		.080	.490	.393	.323	.335	.350
.100	-.106	-.030	-.099	-.061		.130	-.300	-.526	-.596	-.533	-.574
.145	-.087	-.067	-.062	-.037		.145	-4.220	-3.985	-3.688	-4.082	-3.790
.189	-.019	-.012	.000	-.012		.155	-1.507	-1.536	-1.485	-1.414	-1.130
.234	-.044	.067	.000	.018		.180	-1.145	-.986	-1.047	-.906	-1.061
.280	-.050	.085	-.037	.024		.220	-.674	-.701	-.706	-.769	-.674
.326	-.044	.091	-.037	.061		.270	-.521	-.623	-.590	-.502	-.518
.371	-.100	.127	-.074	.146		.400	-.557	-.599	-.450	-.434	-.512
.392	-.037	.132	.012	.347		.620	-1.041	-.949	.140	-.670	-.855
.413	-.150	.139	-.124	-.134		.685	-6.026	-4.572	.055	-1.439	-6.487
.434	-.296	.175	-.354	-.256		.693	-5.941	-5.037	-1.168	-1.700	-5.544
.457	-.237	.195	-.459	-.183		.700	-3.742	-3.538	-1.041	-1.141	-3.859
.480	-.300	.215	-.422	-.049		.720	-1.782	-1.361	-.730	-.751	-1.249
.502	-.368	.235	-.447	.079		.750	-1.194	-.804	-.749	-.788	-.774
.551	-.412	.255	-.608	.274		.800	-.790	-.599	-.785	-.806	-.706
.585	-.406	.284	-.738	.231		.900	-.508	-.635	-.663	-.806	-.662
.592	-.381	.284	-.850	-.852		.980	-.024	-.544	-.657	-.695	-.456
.613	-.287	.212	-.689	.780	Lower	.025	-.251	-.127	.043	.081	-.031
.634	-.244	.169	-.484	-.761		.120	-.312	-.103	.024	.025	-.069
.655	-.212	.103	-.347	-.183		.220	-.276	-.145	-.012	.012	-.044
.675	-.137	.012	-.217	-.085		.300	-.067	-.194	-.073	-.043	-.075
.696	-.075	-.006	-.149	-.037		.620	.521	.381	.110	-.043	-.187
.774	.012	.030	-.056	.024		.750	.790	.605	.110	.068	.081
.852	.012	-.030	.012	-.103		.850	.704	.659	.323	.211	.169
.930	.081	-.151	.087	-.195		.950	.539	.393	.176	.180	.206
$\alpha = 5.8°$											
.032	.108	.496	.141	.318	Upper	.010	.662	.628	.591	.635	.639
.053	-.120	.270	-.064	.084		.080	-.108	-.145	-.182	-.212	-.108
.100	-.183	.069	-.186	-.117		.130	-1.248	-1.401	-1.494	-1.436	-1.417
.145	-.145	.013	-.154	-.084		.145	-6.431	-6.039	-5.607	-6.141	-5.649
.189	-.057	.057	-.103	-.039		.155	-2.611	-2.583	-2.521	-2.462	-2.012
.234	-.108	.132	-.019	-.039		.180	-1.891	-1.647	-1.715	-1.590	-1.702
.280	-.101	.145	.038	-.039		.220	-1.172	-1.169	-1.182	-1.244	-1.132
.326	-.114	.157	.013	-.026		.270	-.891	-.993	-.949	-.872	-.905
.371	-.202	.226	-.154	.019		.400	-.790	-.811	-.663	-.660	-.822
.392	-.250	.260	-.244	.175		.620	-1.159	-1.062	.026	-.737	-1.164
.413	-.297	.289	-.526	.136		.685	-5.578	-4.594	.078	-1.474	-8.483
.434	-.354	.333	-.808	.169		.693	-5.381	-4.933	-1.293	-1.603	-7.654
.457	-.392	.350	-.756	.338		.700	-3.375	-3.526	-1.143	-1.077	-5.693
.480	-.443	.370	-.679	.435		.720	-1.611	-1.326	-.819	-.776	-2.239
.502	-.519	.390	-.654	.455		.750	-1.172	-.830	-.884	-.846	-1.480
.551	-.506	.410	-.744	.494		.800	-.872	-.742	-.897	-.865	-1.158
.585	-.481	.427	-.872	.520		.900	-.548	-.710	-.682	-.859	-.949
.592	-.468	.421	-.949	-.949		.980	-.153	-.660	-.689	-.718	-.430
.613	-.329	.339	-.731	-.637	Lower	.025	-.038	.383	.338	.346	.171
.634	-.304	.258	-.494	-.507		.120	.204	.308	.299	.256	.089
.655	-.266	.207	-.333	-.091		.220	.535	.333	.273	.250	.095
.675	-.164	.031	-.224	-.026		.300	.605	.478	.435	.410	.367
.696	-.127	.019	-.154	.000		.620	.700	.710	.676	.686	.329
.774	-.025	.075	-.019	.052		.750	.815	.804	.747	.737	.633
.852	-.019	.044	.006	-.091		.850	.637	.616	.559	.577	.588
.930	.051	-.113	.083	-.156		.950	.471	.258	.175	.205	.323
$\alpha = 13.3°$											
.032	-.057	.650	-.097	.248	Upper	.010	-.133	-1.068	-1.248	-1.174	-.153
.053	-.223	.438	-.271	.045		.080	-.955	-.955	-1.000	-1.052	-.764
.100	-.166	.212	-.348	-.166		.130	-2.387	-2.553	-2.617	-2.530	-2.401
.145	-.121	.119	-.303	-.172		.145	-8.654	-8.349	-7.399	-8.125	-7.329
.189	-.045	.153	-.252	-.127		.155	-3.607	-3.767	-3.553	-3.569	-2.897
.234	-.089	.212	-.065	-.140		.180	-2.460	-2.414	-2.394	-2.278	-2.248
.280	-.096	.212	.090	-.172		.220	-1.525	-1.698	-1.624	-1.684	-1.477
.326	-.121	.232	.052	-.197		.270	-1.141	-1.340	-1.242	-1.200	-1.127
.371	-.223	.312	-.213	-.197		.400	-.942	-1.028	-.777	-.820	-.962
.392	-.280	.370	-.561	-.159		.620	-1.101	-1.200	-.127	-.832	-1.350
.413	-.350	.424	-.910	.217		.685	-3.336	-4.536	.045	-2.317	-9.068
.434	-.395	.464	-1.207	.522		.693	-2.885	-4.536	-1.197	-1.529	-8.202
.457	-.395	.466	-1.013	.611		.700	-1.824	-3.236	-1.083	-1.065	-6.088
.480	-.439	.468	-.865	.586		.720	-.981	-1.180	-.745	-.742	-2.382
.502	-.478	.470	-.800	.529		.750	-.842	-.749	-.745	-.807	-1.547
.551	-.420	.473	-.845	.567		.800	-.696	-.809	-.764	-.807	-1.165
.585	-.395	.477	-.845	.611		.900	-.511	-.809	-.732	-.832	-1.012
.592	-.376	.471	-.845	-.809		.980	-.418	-.743	-.681	-.716	-.579
.613	-.261	.365	-.703	-.600	Lower	.025	.458	.690	.669	.678	.541
.634	-.248	.252	-.516	-.503		.120	.836	.816	.758	.716	.592
.655	-.223	.119	-.323	-.337		.220	.782	.756	.745	.723	.624
.675	-.134	.000	-.187	-.115		.300	.690	.696	.675	.652	.522
.696	-.096	.020	-.110	.006		.620	.769	.782	.764	.703	.236
.774	.019	.146	.032	.083		.750	.849	.889	.758	.742	.605
.852	-.019	.073	.000	-.038		.850	.643	.650	.541	.594	.503
.930	.038	-.007	.039	-.006		.950	.411	.279	.197	.200	.274

TABLE 21 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 4.0 h_d/c = 2.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 18.9°											
.032	-.172	.705	-.314	.131	Upper	.010	-3.755	-2.812	-3.022	-3.035	-2.858
.053	-.292	.494	-.451	-.046		.080	-1.650	-1.837	-2.597	-2.682	-1.525
.100	-.199	.310	-.517	-.268		.130	-3.047	-2.931	-2.957	-2.911	-3.223
.145	-.159	.217	-.471	-.294		.145	-9.485	-8.713	-7.830	-8.314	-8.700
.189	-.093	.257	-.425	-.249		.155	-4.035	-4.254	-4.121	-4.042	-3.753
.234	-.066	.283	-.111	-.288		.180	-2.677	-2.812	-2.839	-2.721	-2.977
.280	-.113	.310	.092	-.327		.220	-1.644	-1.982	-1.975	-1.962	-2.042
.326	-.146	.316	.072	-.392		.270	-1.208	-1.541	-1.511	-1.446	-1.645
.371	-.318	.415	-.288	-.425		.400	-.910	-1.100	-.903	-.935	-1.366
.392	-.390	.465	-.837	-.523		.620	-.975	-1.146	-.281	-.798	-1.485
.413	-.464	.533	-1.099	.183		.685	-3.274	-3.234	.000	-1.341	-8.210
.434	-.517	.560	-1.675	.608		.693	-3.125	-3.458	-1.341	-1.557	-7.301
.457	-.511	.550	-1.347	.687		.700	-1.982	-2.430	-1.184	-1.138	-5.358
.480	-.497	.540	-1.066	.648		.720	-.988	-.968	-.831	-.798	-2.003
.502	-.524	.530	-.975	.602		.750	-.760	-.751	-.824	-.824	-1.260
.551	-.431	.520	-.948	.582		.800	-.650	-.836	-.824	-.857	-.948
.585	-.385	.507	-1.007	.628		.900	-.507	-.836	-.778	-.798	-.902
.592	-.345	.494	-1.171	-1.125		.980	-.429	-.810	-.765	-.772	-.623
.613	-.239	.375	-1.001	-.700							
.634	-.239	.270	-.661	-.530	Lower	.025	.702	.836	.811	.791	.650
.655	-.199	.138	-.379	-.406		.120	.884	.850	.752	.726	.610
.675	-.119	.020	-.222	-.177		.220	.832	.797	.798	.752	.630
.696	-.066	.053	-.092	-.007		.300	.741	.744	.687	.674	.570
.774	.027	.204	.039	.105		.620	.786	.803	.772	.726	.239
.852	-.066	.099	-.039	-.007		.750	.851	.863	.791	.759	.603
.930	.	.092	.	.078		.850	.676	.665	.569	.589	.504
						.950	.442	.349	.229	.242	.279
α = 23.0°											
.032	-.239	.808	-.464	.040	Upper	.010	-8.307	-3.799	-3.926	-3.772	-3.442
.053	-.345	.621	-.585	-.127		.080	-2.043	-3.512	-4.020	-3.806	-2.089
.100	-.199	.387	-.639	-.394		.130	-3.379	-2.945	-3.158	-2.972	-2.314
.145	-.166	.300	-.585	-.407		.145	-9.896	-8.180	-6.637	-6.509	-6.107
.189	-.080	.321	-.545	-.347		.155	-4.193	-4.294	-3.839	-3.463	-2.533
.234	-.066	.341	-.134	-.427		.180	-2.771	-2.958	-2.798	-2.448	-2.016
.280	-.113	.361	.141	-.447		.220	-1.696	-2.103	-1.956	-1.742	-1.253
.326	-.199	.367	.013	-.548		.270	-1.376	-1.636	-1.502	-1.318	-.928
.371	-.411	.474	-.437	-.628		.400	-1.035	-1.189	-1.002	-.948	-.869
.392	-.500	.530	-.988	-.808		.620	-1.002	-1.142	-.815	-.827	-.862
.413	-.584	.594	-1.291	.174		.685	-4.060	-2.597	-.434	-1.977	-1.737
.434	-.610	.614	-2.078	.648		.693	-4.080	-2.798	-1.810	-2.152	-1.393
.457	-.557	.600	-1.560	.721		.700	-2.718	-2.063	-1.643	-1.399	-1.187
.480	-.491	.580	-1.311	.681		.720	-1.429	-.948	-1.202	-.888	-.822
.502	-.471	.560	-1.123	.668		.750	-1.002	-.821	-1.082	-.894	-.849
.551	-.305	.540	-1.069	.601		.800	-.708	-.821	-.875	-.834	-.836
.585	-.252	.521	-1.123	.654		.900	-.481	-.848	-.808	-.800	-.809
.592	-.225	.521	-1.473	-1.396		.980	-.374	-.735	-.795	-.760	-.756
.613	-.126	.381	-1.298	-.401							
.634	-.146	.260	-.787	-.641	Lower	.025	.801	.881	.841	.854	.690
.655	-.133	.127	-.437	-.361		.120	.901	.875	.788	.787	.623
.675	-.080	-.007	-.256	-.114		.220	.868	.828	.828	.793	.676
.696	-.040	.033	-.141	-.013		.300	.775	.795	.741	.753	.610
.774	-.013	.180	.007	.100		.620	.801	.828	.788	.767	.351
.852	-.119	.087	-.061	-.027		.750	.875	.861	.815	.773	.617
.930	-.046	.114	-.020	.127		.850	.694	.688	.574	.632	.504
						.950	.474	.381	.240	.276	.212

L-927

L-927

TABLE 21 Continued (d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 6.0$ $h_d/c = 3.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
$\alpha = -1.4°$											
.032	.271	.304	.298	.285	Upper	.010	.955	.876	.844	.834	.820
.053	.068	.061	.079	.050		.080	.490	.389	.335	.335	.370
.100	-.099	-.049	-.103	-.093		.130	-.316	-.505	-.571	-.505	-.505
.145	-.080	-.067	-.067	-.074		.145	-4.187	-3.937	-3.598	-3.810	-3.501
.189	[illegible]	-.012	.006	-.012		.155	-1.520	-1.528	-1.445	-1.278	-1.042
.234	-.049	.055	.006	.012		.180	-1.154	-.962	-.986	-.797	-.943
.280	-.049	.079	-.043	.025		.220	-.670	-.682	-.633	-.639	-.586
.326	-.025	.091	-.018	.062		.270	-.533	-.584	-.496	-.389	-.438
.371	-.092	.122	-.055	.118		.400	-.571	-.536	-.292	-.262	-.431
.392	-.037	.128	.037	.304		.620	-1.030	-.889	.509	-.676	-.777
.413	-.129	.134	-.116	-.248		.685	-5.931	-4.217	-.087	-1.661	-6.355
.434	-.191	.158	-.335	-.335		.693	-5.856	-4.631	-1.203	-1.844	-5.535
.457	-.210	.165	-.432	-.199		.700	-3.710	-3.250	-1.079	-1.351	-3.877
.480	-.271	.185	-.383	-.068		.720	-1.774	-1.193	-.763	-.761	-1.208
.502	-.357	.215	-.396	.099		.750	-1.191	-.724	-.794	-.724	-.703
.551	-.394	.240	-.566	.248		.800	-.794	-.651	-.794	-.712	-.598
.585	-.382	.274	-.688	.217		.900	-.527	-.609	-.682	-.736	-.598
.592	-.357	.280	-.791	-.825		.980	-.043	-.548	-.670	-.645	-.388
.613	-.284	.225	-.633	-.658							
.634	-.247	.170	-.444	-.682	Lower	.025	-.323	-.158	-.006	.073	-.025
.655	-.197	.097	-.292	-.211		.120	-.372	-.134	-.012	.030	-.055
.675	-.123	.012	-.207	-.099		.220	-.298	-.164	-.037	.018	-.055
.696	-.074	[illegible]	-.140	-.062		.300	-.037	-.201	-.105	-.055	-.080
.774	[illegible]	.055	-.043	.025		.620	.558	.396	.099	-.006	-.203
.852	[illegible]	-.037	.030	-.105		.750	.794	.621	.112	.103	.049
.930	.080	-.176	.097	-.174		.850	.682	.663	.360	.231	.173
						.950	.527	.353	.211	.189	.210
$\alpha = 5.8°$											
.032	.084	.471	.114	.297	Upper	.010	.663	.641	.633	.645	.669
.053	-.123	.249	-.063	.070		.080	-.104	-.157	-.133	-.145	-.065
.100	-.214	.072	-.202	-.114		.130	-1.208	-1.400	-1.366	-1.290	-1.325
.145	-.136	-.007	-.164	-.089		.145	-6.347	-5.992	-5.225	-5.630	-5.347
.189	-.071	.052	-.095	-.044		.155	-2.566	-2.564	-2.328	-2.214	-1.923
.234	-.104	.131	-.025	-.038		.180	-1.865	-1.635	-1.556	-1.404	-1.585
.280	-.104	.118	.025	-.032		.220	-1.130	-1.145	-1.037	-1.063	-1.052
.326	-.084	.131	.006	-.019		.270	-.864	-.942	-.784	-.702	-.806
.371	-.182	.183	-.120	.019		.400	-.741	-.765	-.418	-.424	-.741
.392	-.230	.225	-.215	.171		.620	-1.104	-.962	.481	-.715	-1.072
.413	-.286	.275	-.506	.101		.685	-5.509	-4.108	.044	-1.714	-8.154
.434	-.338	.314	-.772	.183		.693	-5.295	-4.441	-1.208	-1.936	-7.400
.457	-.377	.335	-.727	.316		.700	-3.261	-3.127	-1.063	-1.468	-5.509
.480	-.409	.355	-.645	.405		.720	-1.572	-1.125	-.753	-.841	-2.092
.502	-.494	.375	-.614	.411		.750	-1.124	-.720	-.797	-.803	-1.345
.551	-.481	.395	-.702	.462		.800	-.832	-.746	-.822	-.791	-1.014
.585	-.461	.412	-.822	.481		.900	-.565	-.680	-.709	-.797	-.812
.592	-.448	.406	-.898	-.898		.980	-.136	-.674	-.696	-.677	-.364
.613	-.325	.327	-.683	-.658							
.634	-.299	.242	-.468	-.493	Lower	.025	-.091	.327	.304	.335	.143
.655	-.266	.118	-.329	-.127		.120	.208	.262	.278	.253	.065
.675	-.175	-.007	-.228	-.051		.220	.565	.294	.266	.215	.071
.696	-.117	-.020	-.152	-.051		.300	.611	.451	.430	.335	.299
.774	.006	.072	-.057	.013		.620	.702	.693	.658	.683	.338
.852	-.026	.026	[illegible]	-.114		.750	.832	.772	.721	.746	.630
.930	.052	-.137	.057	-.171		.850	.637	.582	.531	.595	.578
						.950	.468	.203	.158	.221	.357
$\alpha = 13.4°$											
.032	-.105	.643	-.104	.262	Upper	.010	-.078	-.806	-1.145	-1.027	-.151
.053	-.250	.448	-.266	.052		.080	-.877	-.845	-.975	-.962	-.777
.100	-.204	.240	-.357	-.170		.130	-2.276	-2.358	-2.531	-2.371	-2.384
.145	-.138	.149	-.305	-.150		.145	-8.379	-7.913	-7.136	-7.614	-7.205
.189	-.086	.175	-.260	-.111		.155	-3.493	-3.547	-3.454	-3.320	-2.865
.234	-.119	.240	-.065	-.150		.180	-2.348	-2.235	-2.270	-2.085	-2.200
.280	-.119	.240	.078	-.157		.220	-1.478	-1.553	-1.518	-1.501	-1.436
.326	-.138	.247	.045	-.203		.270	-1.099	-1.221	-1.125	-1.027	-1.093
.371	-.263	.338	-.221	-.203		.400	-.896	-.858	-.556	-.572	-.922
.392	-.310	.390	-.565	-.150		.620	-1.125	-1.001	.059	-.767	-1.311
.413	-.375	.435	-.890	.196		.685	-4.271	-3.963	-.425	-1.488	-8.983
.434	-.441	.474	-1.195	.530		.693	-3.800	-3.703	-1.177	-1.332	-8.160
.457	-.428	.476	-.981	.602		.700	-2.459	-2.612	-1.053	-.916	-6.085
.480	-.448	.478	-.838	.556		.720	-1.347	-.871	-.726	-.682	-2.364
.502	-.514	.480	-.773	.530		.750	-1.027	-.734	-.733	-.741	-1.521
.551	-.461	.480	-.838	.556		.800	-.739	-.728	-.746	-.760	-1.093
.585	-.441	.481	-.884	.615		.900	-.399	-.682	-.713	-.806	-.889
.592	-.408	.474	-.916	-.870		.980	-.222	-.643	-.654	-.676	-.474
.613	-.290	.377	-.721	-.477							
.634	-.263	.273	-.487	-.556	Lower	.025	.445	.682	.661	.656	.527
.655	-.250	.175	-.318	-.105		.120	.785	.819	.746	.728	.586
.675	-.158	.039	-.201	-.046		.220	.759	.747	.726	.721	.626
.696	-.119	.045	-.143	-.033		.300	.667	.695	.667	.630	.514
.774	-.013	.143	-.013	.033		.620	.739	.799	.759	.708	.231
.852	-.053	.091	-.032	-.059		.750	.831	.890	.778	.741	.606
.930	.020	-.006	.039	-.033		.850	.634	.650	.536	.585	.494
						.950	.451	.260	.183	.208	.283

TABLE 21 Concluded
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = .0 h_d/c = 3.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :									
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.425	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
					α = 19.0°						
.032	-.184	.729	-.310	.137	Upper	.010	-3.700	-2.549	-2.832	-2.852	-2.489
.053	-.296	.523	-.461	-.033		.080	-1.671	-1.451	-2.342	-2.397	-1.304
.100	-.191	.303	-.514	-.268		.130	-3.090	-2.807	-2.872	-2.766	-3.036
.145	-.138	.213	-.468	-.294		.145	-9.556	-8.409	-7.529	-7.969	-8.153
.189	-.072	.245	-.421	-.242		.155	-4.072	-4.046	-3.931	-3.846	-3.517
.234	-.066	.290	-.119	-.307		.180	-2.706	-2.619	-2.662	-2.549	-2.707
.280	-.099	.303	.099	-.307		.220	-1.658	-1.819	-1.805	-1.785	-1.824
.326	-.132	.316	.072	-.392		.270	-1.233	-1.400	-1.328	-1.264	-1.423
.371	-.329	.400	-.296	-.438		.400	-.915	-.902	-.641	-.692	-1.185
.392	-.400	.455	-.803	-.517		.620	-1.081	-.994	-.255	-.784	-1.330
.413	-.461	.516	-1.054	.170		.685	-4.072	-2.497	-.242	-1.449	-7.705
.434	-.514	.549	-1.594	.615		.693	-3.846	-2.220	-1.190	-1.390	-6.849
.457	-.501	.535	-1.291	.700		.700	-2.507	-1.581	-1.079	-1.027	-4.966
.480	-.481	.525	-1.014	.641		.720	-1.359	-.787	-.752	-.711	-1.739
.502	-.487	.515	-.922	.621		.750	-1.001	-.832	-.759	-.757	-1.008
.551	-.369	.500	-.935	.589		.800	-.736	-.800	-.746	-.764	-.711
.585	-.316	.510	-1.047	.634		.900	-.484	-.749	-.752	-.803	-.790
.592	-.310	.484	-1.324	-1.276		.980	-.305	-.729	-.700	-.731	-.626
.613	-.204	.381	-1.054	-.720							
.634	-.198	.265	-.606	-.628	Lower	.025	.683	.820	.805	.777	.659
.655	-.191	.136	-.342	-.242		.120	.862	.813	.752	.698	.566
.675	-.099	.013	-.184	-.065		.220	.809	.755	.746	.738	.619
.696	-.066	.045	-.119	-.033		.300	.743	.749	.687	.678	.560
.774	.013	.161	[illegible]	.078		.620	.782	.807	.765	.711	.263
.852	-.053	.090	-.046	-.026		.750	.836	.858	.778	.738	.606
.930	.020	.084	-.007	.072		.850	.663	.652	.569	.599	.527
						.950	.464	.277	.222	.237	.277
					α = 23.1°						
.032	-.125	.857	-.467	.067	Upper	.010	-8.643	-4.143	-3.800	-3.600	-3.125
.053	-.313	.571	-.533	-.067		.080	-2.143	-3.929	-3.800	-3.600	-1.688
.100	-.125	.357	-.533	-.400		.130	-3.714	-3.429	-2.933	-2.867	-2.188
.145	-.125	.286	-.533	-.467		.145	-10.571	-8.643	-6.467	-6.533	-3.750
.189	-.063	.214	-.533	-.400		.155	-4.571	-4.714	-3.667	-3.400	-2.188
.234	-.063	.286	[illegible]	-.467		.180	-3.000	-3.071	-2.600	-2.400	-1.750
.280	-.125	.286	.133	-.467		.220	-1.857	-2.214	-1.733	-1.600	-1.063
.326	-.188	.357	.133	-.600		.270	-1.500	-1.786	-1.267	-1.200	-.750
.371	-.375	.500	-.400	-.667		.400	-1.214	-1.286	-.800	-.867	-.750
.392	-.438	.572	-1.067	-.867		.620	-1.214	-1.143	-.600	-.933	-.750
.413	-.500	.643	-1.333	.200		.685	-4.143	-1.286	-.333	-2.333	-1.500
.434	-.500	.643	-2.000	.733		.693	-4.357	-1.429	-1.600	-2.667	-1.188
.457	-.438	.621	-1.533	.800		.700	-2.857	-1.357	-1.467	-1.533	-1.000
.480	-.438	.599	-1.200	.733		.720	-1.571	-1.143	-1.133	-1.000	-.688
.502	-.438	.577	-1.067	.667		.750	-1.214	-1.071	-1.133	-.933	-.688
.551	-.313	.533	-1.067	.600		.800	-1.000	-1.071	-1.000	-.867	-.688
.585	-.250	.500	-1.133	.600		.900	-.714	-1.000	-.867	-.800	-.688
.592	-.125	.500	-1.400	-1.400		.980	-.500	-1.000	-.800	-.733	-.625
.613	-.125	.357	-1.400	-.875							
.634	-.125	.143	-1.000	-.800	Lower	.025	.857	.857	.867	.867	.750
.655	-.125	[illegible]	-.533	-.267		.120	.929	.857	.800	.800	.688
.675	-.063	-.143	-.333	-.067		.220	.929	.857	.800	.800	.688
.696	[illegible]	-.143	-.200	-.067		.300	.857	.786	.800	.800	.625
.774	-.032	-.071	[illegible]	-.067		.620	.857	.857	.867	.800	.375
.852	-.063	[illegible]	.000	-.067		.750	-.929	.857	.867	.800	.688
.930	.000	.071	[illegible]	.200		.850	.786	.714	.600	.667	.563
						.950	.571	.286	.200	.267	.250

L-927

L-927

TABLE 22(a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 8.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.6°											
.032	.285	.308	.289	.310	Upper	.010	.955	.863	.837	.804	.813
.053	.050	.068	.050	.062		.080	.443	.302	.279	.233	.347
.100	-.099	-.043	-.101	-.074		.130	-.411	-.647	-.769	-.716	-.633
.145	-.093	-.086	-.088	-.050		.145	-4.567	-4.321	-4.125	-4.481	-3.933
.189	-.012	-.012	-.025	.012		.155	-1.683	-1.744	-1.749	-1.672	-1.266
.234	-.062	.062	-.038	.012		.180	-1.278	-1.146	-1.253	-1.100	-1.104
.280	-.056	.068	-.063	.037		.220	-.772	-.851	-.893	-.899	-.713
.326	-.043	.086	-.019	.081		.270	-.614	-.777	-.794	-.641	-.515
.371	-.130	.142	-.063	.155		.400	-.696	-.789	-.757	-.515	-.515
.392	-.143	.151	.013	.366		.620	-1.227	-1.368	-1.017	-.836	-.893
.413	-.155	.160	-.176	-.025		.685	-6.630	-6.201	-3.759	-2.105	-6.197
.434	-.223	.185	-.415	-.180		.693	-6.579	-6.792	-4.876	-2.262	-5.230
.457	-.261	.203	-.522	-.099		.700	-4.200	-4.851	-3.511	-1.691	-3.672
.480	-.335	.221	-.490	.012		.720	-2.018	-2.083	-1.259	-1.018	-1.241
.502	-.409	.239	-.522	.167		.750	-1.385	-1.282	-.868	-1.050	-.844
.551	-.447	.275	-.742	.310		.800	-.936	-.752	-.713	-.924	-.788
.585	-.434	.302	-.899	.254		.900	-.557	-.271	-.614	-.823	-.701
.592	-.422	.302	-1.037	-.931		.980	.013	.117	-.565	-.773	-.527
.613	-.329	.234	-.855	-.850							
.634	-.273	.173	-.603	-.782	Lower	.025	-.183	-.074	.099	.113	.025
.655	-.211	.111	-.427	-.186		.120	-.259	-.037	.074	.069	.000
.675	-.118	.031	-.277	-.062		.220	-.228	-.068	.037	.050	.000
.696	-.056	.018	-.201	-.043		.300	-.051	-.099	-.043	-.013	-.037
.774	-.037	.062	-.088	.019		.620	.519	.382	.174	.025	-.248
.852	-.025	-.049	-.025	-.136		.750	.791	.573	.149	.157	.118
.930	.081	-.222	.082	-.304		.850	.727	.703	.422	.264	.248
						.950	.557	.524	.273	.201	.254
α = 5.6°											
.032	.089	.471	.146	.336	Upper	.010	.576	.547	.516	.554	.586
.053	-.115	.251	-.070	.071		.080	-.229	-.264	-.297	-.306	-.178
.100	-.178	.069	-.204	-.097		.130	-1.393	-1.540	-1.691	-1.560	-1.484
.145	-.134	-.025	-.166	-.084		.145	-6.829	-6.234	-6.027	-6.330	-5.763
.189	-.057	.044	-.108	-.032		.155	-2.806	-2.715	-2.762	-2.643	-2.165
.234	-.089	.113	-.045	-.039		.180	-2.054	-1.772	-1.942	-1.732	-1.745
.280	-.076	.113	.038	-.032		.220	-1.289	-1.307	-1.387	-1.350	-1.178
.326	-.115	.126	[illegible]	-.045		.270	-1.001	-1.106	-1.181	-.981	-.949
.371	-.185	.189	-.166	.006		.400	-.883	-.980	-1.026	-.713	-.853
.392	-.242	.236	-.280	.168		.620	-1.151	-1.401	-1.174	-.911	-1.204
.413	-.299	.283	-.548	.200		.685	-4.180	-6.083	-3.711	-1.859	-8.367
.434	-.350	.314	-.866	.226		.693	-3.833	-6.567	-4.795	-2.063	-7.559
.457	-.382	.330	-.783	.348		.700	-2.335	-4.688	-3.401	-1.566	-5.642
.480	-.439	.347	-.726	.445		.720	-.981	-2.011	-1.316	-1.114	-2.286
.502	-.503	.363	-.688	.432		.750	-.831	-1.219	-1.033	-1.184	-1.547
.551	-.503	.396	-.764	.478		.800	-.759	-.704	-.891	-1.070	-1.223
.585	-.478	.421	-.841	.542		.900	-.628	-.314	-.716	-.891	-1.070
.592	-.439	.402	-.860	-.858		.980	-.451	-.019	-.678	-.898	-.599
.613	-.337	.308	-.669	-.700							
.634	-.312	.233	-.509	-.613	Lower	.025	.157	.377	.381	.388	.197
.655	-.261	.132	-.350	-.439		.120	.216	.314	.303	.242	.096
.675	-.178	.025	-.223	-.136		.220	.536	.427	.361	.344	.197
.696	-.121	.013	-.121	-.006		.300	.595	.572	.574	.541	.458
.774	-.025	.075	.006	-.071		.620	.687	.691	.671	.624	.248
.852	-.057	.044	-.013	-.136		.750	.831	.754	.671	.656	.605
.930	.057	-.107	.064	-.148		.850	.602	.635	.561	.535	.497
						.950	.379	.465	.200	.166	.267
α = 13.2°											
.032	-.071	.652	-.083	.256	Upper	.010	-.242	-1.475	-1.551	-1.516	-1.128
.053	-.237	.435	-.274	.032		.080	-.955	-1.027	-1.115	-1.146	-.974
.100	-.186	.204	-.363	-.179		.130	-2.362	-2.641	-2.840	-2.668	-2.673
.145	-.128	.138	-.318	-.160		.145	-8.323	-8.502	-7.897	-8.291	-7.833
.189	-.077	.132	-.255	-.128		.155	-3.502	-3.899	-3.904	-3.770	-3.256
.234	-.077	.211	-.083	-.167		.180	-2.382	-2.529	-2.660	-2.433	-2.462
.280	-.122	.217	.076	-.179		.220	-1.535	-1.811	-1.878	-1.815	-1.660
.326	-.128	.217	.032	-.224		.270	-1.153	-1.488	-1.513	-1.337	-1.301
.371	-.269	.323	-.229	-.199		.400	-.993	-1.159	-1.167	-.891	-1.122
.392	-.314	.376	-.592	-.199		.620	-1.076	-1.488	-1.250	-1.025	-1.500
.413	-.359	.428	-.949	.199		.685	-3.050	-5.940	-3.026	-3.598	-9.135
.434	-.423	.474	-1.280	.526		.693	-2.356	-6.039	-3.596	-2.019	-8.250
.457	-.436	.473	-1.025	.615		.700	-1.490	-4.314	-2.468	-1.503	-6.147
.480	-.449	.472	-.885	.558		.720	-.650	-1.791	-1.237	-1.076	-2.500
.502	-.538	.471	-.834	.526		.750	-.567	-1.034	-1.090	-1.159	-1.692
.551	-.468	.469	-.885	.564		.800	-.541	-.560	-.974	-1.140	-1.359
.585	-.429	.468	-.872	.641		.900	-.446	-.237	-.782	-.936	-1.288
.592	-.404	.468	-.866	-.859		.980	-.490	-.198	-.808	-.936	-.897
.613	-.327	.362	-.707	-.550							
.634	-.276	.257	-.509	-.404	Lower	.025	.471	.705	.705	.681	.551
.655	-.256	.132	-.344	-.404		.120	.790	.803	.756	.694	.558
.675	-.160	-.007	-.210	-.269		.220	.745	.731	.744	.694	.603
.696	-.103	.033	-.102	-.071		.300	.662	.692	.679	.643	.506
.774	-.090	.165	-.060	-.077		.620	.720	.790	.718	.650	.179
.852	-.077	.053	-.019	-.083		.750	.821	.902	.731	.707	.571
.930	.019	[illegible]	.006	.000		.850	.605	.692	.545	.541	.449
						.950	.363	.494	.192	.159	.186

TABLE 22 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 8.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 18.8°											
.032	-.179	.748	-.314	.141	Upper	.010	-4.932	-3.118	-3.456	-3.342	-3.117
.053	-.305	.521	-.445	-.054		.080	-1.786	-2.083	-3.228	-3.231	-1.684
.100	-.212	.334	-.530	-.276		.130	-3.153	-3.005	-3.127	-3.081	-3.203
.145	-.172	.240	-.497	-.343		.145	-9.733	-8.974	-8.237	-7.993	-8.634
.189	-.086	.240	-.432	-.276		.155	-4.173	-4.454	-4.451	-4.180	-3.846
.234	-.099	.287	-.118	-.336		.180	-2.754	-2.951	-3.147	-2.885	-2.991
.280	-.133	.314	.098	-.343		.220	-1.707	-2.110	-2.239	-2.113	-2.076
.326	-.172	.314	.033	-.451		.270	-1.295	-1.696	-1.789	-1.648	-1.698
.371	-.338	.441	-.340	-.511		.400	-.988	-1.269	-1.325	-1.164	-1.558
.392	-.411	.495	-.850	-.639		.620	-1.020	-1.489	-1.257	-1.465	-1.545
.413	-.484	.548	-1.112	.161		.685	-3.087	-4.661	-1.876	-2.858	-6.353
.434	-.544	.581	-1.760	.619		.693	-2.885	-4.901	-2.817	-3.055	-5.650
.457	-.531	.575	-1.361	.686		.700	-1.786	-3.526	-2.212	-2.015	-4.377
.480	-.511	.569	-1.125	.666		.720	-.877	-1.496	-1.264	-1.276	-1.976
.502	-.537	.563	-1.047	.612		.750	-.628	-.881	-1.123	-1.262	-1.492
.551	-.444	.551	-1.001	.578		.800	-.517	-.507	-1.036	-1.132	-1.273
.585	-.411	.541	-1.105	.652		.900	-.419	-.294	-.921	-.994	-1.200
.592	-.398	.514	-1.276	-1.224		.980	-.477	-.220	-.928	-.981	-.968
.613	-.272	.407	-.955	-.600	Lower	.025	.693	.868	.834	.791	.663
.634	-.252	.287	-.621	-.424		.120	.890	.841	.780	.752	.564
.655	-.212	.154	-.360	-.417		.220	.824	.801	.800	.759	.610
.675	-.106	.027	-.209	-.262		.300	.746	.755	.733	.700	.544
.696	-.080	.013	-.078	-.081		.620	.759	.808	.767	.667	.139
.774	-.087	.043	.046	-.061		.750	.857	.888	.767	.706	.577
.852	-.093	.073	-.052	-.040		.850	.661	.721	.598	.556	.451
.930	.007	.093	-.039	.087		.950	.432	.494	.215	.209	.199
α = 23.0°											
.032	-.252	.824	-.407		Upper	.010	-8.833	-3.885	-5.000	-3.769	-3.667
.053	-.365	.654	-.516	-.154		.080	-2.016	-3.552	-5.077	-3.782	-2.341
.100	-.199	.406	-.574	-.462		.130	-3.329	-2.826	-4.000	-3.123	-2.420
.145	-.179	.314	-.555	-.538		.145	-9.682	-8.104	-7.769	-5.860	-6.452
.189	-.073	.327	-.503	-.462		.155	-4.105	-4.258	-4.615	-3.349	-2.858
.234	-.066	.366	-.116	-.538		.180	-2.692	-2.917	-3.538	-2.426	-2.182
.280	-.119	.366	.097	-.615		.220	-1.691	-2.106	-2.538	-1.788	-1.399
.326	-.206	.392	.006	-.692		.270	-1.432	-1.661	-2.000	-1.420	-1.054
.371	-.431	.497	-.445	-.846		.400	-1.048	-1.230	-1.538	-1.071	-1.008
.392	-.517	.556	-.910	-1.077		.620	-.995	-1.204	-1.308	-1.284	-.968
.413	-.603	.615	-1.207	.154		.685	-3.800	-2.924	-1.154	-1.813	-1.837
.434	-.637	.621	-1.968	.769		.693	-3.972	-3.349	-2.308	-1.897	-1.519
.457	-.584	.610	-1.471	.769		.700	-2.613	-2.401	-2.000	-1.329	-1.320
.480	-.497	.599	-1.265	.692		.720	-1.353	-1.014	-1.308	-.942	-.962
.502	-.497	.588	-1.091	.692		.750	-.902	-.654	-1.308	-.929	-.975
.551	-.338	.566	-1.033	.615		.800	-.643	-.510	-1.231	-.878	-.968
.585	-.245	.549	-1.110	.615		.900	-.484	-.432	-1.154	-.839	-.962
.592	-.245	.510	-1.446	-1.769		.980	-.491	-.340	-1.077	-.800	-.895
.613	-.166	.406	-1.187	-.975	Lower	.025	.802	.883	.923	.820	.683
.634	-.146	.275	-.736	-.846		.120	.902	.837	.846	.781	.610
.655	-.153	.150	-.400	-.538		.220	.862	.824	.846	.768	.663
.675	-.060	.020	-.207	-.154		.300	.796	.798	.769	.755	.590
.696	-.033	.059	-.090	-.077		.620	.782	.911	.769	.710	.245
.774	-.033	.069	-.078	-.038		.750	.882	.890	.769	.723	.603
.852	-.126	.078	-.065	.000		.850	.676	.726	.615	.594	.484
.930	-.033	.157	-.019	.154		.950	.484	.464	.231	.245	.179

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TABLE 22 Continued
(b)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 6.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -1.6°											
.032	.304	.298	.300	.298	Upper	.010	.949	.868	.837	.824	.816
.053	.057	.087	.062	.068		.080	.436	.304	.261	.244	.348
.100	-.101	-.043	-.087	-.074		.130	-.430	-.658	-.794	-.712	-.639
.145	-.108	-.068	-.094	-.062		.145	-4.599	-4.318	-4.200	-4.577	-3.973
.189	-.013	-.012	-.012	-.006		.155	-1.708	-1.731	-1.749	-1.673	-1.278
.234	-.038	.068	-.025	.012		.180	-1.284	-1.117	-1.247	-1.124	-1.113
.280	-.051	.087	-.044	.025		.220	-.791	-.844	-.912	-.930	-.721
.326	-.044	.087	-.037	.062		.270	-.639	-.763	-.806	-.674	-.569
.371	-.120	.136	-.119	.143		.400	-.671	-.775	-.806	-.612	-.563
.392	-.142	.158	-.006	.347		.620	-1.221	-1.359	-1.067	-.780	-.911
.413	-.164	.180	-.175	-.050		.685	-6.585	-6.266	-4.100	-1.879	-6.060
.434	-.221	.192	-.418	-.186		.693	-6.535	-6.842	-5.372	-2.117	-5.048
.457	-.253	.210	-.518	-.074		.700	-4.131	-4.888	-3.927	-1.486	-3.593
.480	-.335	.229	-.493	.062		.720	-1.999	-2.122	-1.427	-.987	-1.240
.502	-.411	.247	-.518	.186		.750	-1.341	-1.303	-.881	-1.105	-.886
.551	-.424	.283	-.687	.347		.800	-.905	-.757	-.720	-.899	-.822
.585	-.424	.310	-.868	.323		.900	-.563	-.248	-.639	-.837	-.709
.592	-.418	.273	-1.011	-.968		.980	.000	.155	-.558	-.768	-.544
.613	-.323	.236	-.849	-.800	Lower	.025	-.145	-.043	.112	.156	.019
.634	-.266	.186	-.624	-.757		.120	-.209	-.012	.099	.125	-.019
.655	-.221	.118	-.425	-.167		.220	-.164	-.037	.062	.081	.006
.675	-.120	.025	-.300	-.062		.300	.006	-.087	[illegible]	.019	-.057
.696	-.063	.012	-.206	-.037		.620	.487	.372	.186	.100	-.196
.774	-.063	-.016	-.081	[illegible]		.750	.765	.583	.155	.162	.120
.852	-.025	-.043	-.019	-.136		.850	.734	.707	.397	.256	.228
.930	.076	-.211	.094	-.279		.950	.563	.596	.279	.212	.190
α = 5.6°											
.032	.077	.465	.141	.316	Upper	.010	.550	.536	.490	.545	.577
.053	-.103	.207	-.083	.045		.080	-.209	-.271	-.336	-.327	-.205
.100	-.192	.065	-.192	-.097		.130	-1.360	-1.613	-1.704	-1.609	-1.519
.145	-.147	-.032	-.179	-.097		.145	-6.655	-6.498	-6.092	-6.513	-6.013
.189	-.077	.039	-.122	-.045		.155	-2.720	-2.846	-2.820	-2.712	-2.263
.234	-.115	.097	-.038	-.052		.180	-1.980	-1.859	-1.955	-1.795	-1.821
.280	-.103	.123	.038	-.039		.220	-1.246	-1.375	-1.407	-1.391	-1.250
.326	-.115	.142	.019	-.039		.270	-.955	-1.162	-1.181	-1.038	-.981
.371	-.224	.200	-.167	-.006		.400	-.841	-1.033	-1.065	-.859	-.904
.392	-.276	.249	-.288	.168		.620	-1.075	-1.484	-1.220	-.859	-1.282
.413	-.327	.297	-.545	.194		.685	-3.909	-6.266	-4.130	-2.205	-8.737
.434	-.372	.323	-.853	.213		.693	-3.549	-6.705	-5.311	-2.141	-7.865
.457	-.410	.339	-.769	.336		.700	-2.069	-4.827	-3.840	-1.571	-5.942
.480	-.455	.355	-.705	.394		.720	-.898	-2.072	-1.484	-1.096	-2.449
.502	-.513	.371	-.673	.407		.750	-.759	-1.265	-1.000	-1.199	-1.692
.551	-.513	.403	-.782	.478		.800	-.702	-.761	-.884	-1.064	-1.372
.585	-.487	.426	-.833	.497		.900	-.550	-.336	-.768	-.897	-1.179
.592	-.455	.336	-.833	-.871		.980	-.487	-.058	-.697	-.865	-.679
.613	-.359	.258	-.647	-.700	Lower	.025	.158	.407	.400	.397	.179
.634	-.308	.207	-.513	-.620		.120	.221	.348	.329	.276	.090
.655	-.250	.123	-.353	-.458		.220	.531	.445	.426	.321	.231
.675	-.167	-.013	-.231	-.161		.300	.620	.568	.613	.577	.468
.696	-.096	.006	-.135	.006		.620	.696	.697	.671	.654	.256
.774	-.086	.084	.013	.006		.750	.810	.768	.684	.705	.615
.852	-.077	.032	-.026	-.129		.850	.607	.645	.561	.545	.474
.930	.038	-.129	.051	-.168		.950	.380	.458	.207	.154	.231
α = 13.2°											
.032	-.071	.595	-.090	.244	Upper	.010	-.345	-1.583	-1.673	-1.577	-1.104
.053	-.258	.340	-.269	-.007		.080	-1.077	-1.027	-1.126	-1.135	-.987
.100	-.200	.177	-.359	-.171		.130	-2.607	-2.695	-2.957	-2.724	-2.756
.145	-.136	.092	-.314	-.191		.145	-9.026	-8.575	-8.226	-8.481	-8.041
.189	-.090	.124	-.269	-.158		.155	-3.846	-3.970	-4.063	-3.891	-3.323
.234	-.103	.150	-.090	-.165		.180	-2.607	-2.571	-2.773	-2.532	-2.536
.280	-.110	.203	.083	-.165		.220	-1.679	-1.825	-1.956	-1.872	-1.717
.326	-.136	.229	.026	-.217		.270	-1.280	-1.465	-1.600	-1.410	-1.329
.371	-.271	.294	-.224	-.237		.400	-1.077	-1.164	-1.245	-1.019	-1.168
.392	-.320	.360	-.596	-.224		.620	-1.178	-1.498	-1.383	-1.058	-1.562
.413	-.368	.425	-.936	.151		.685	-3.541	-6.057	-4.301	-4.519	-9.448
.434	-.426	.464	-1.288	.507		.693	-2.485	-5.959	-4.663	-2.250	-8.505
.457	-.432	.469	-1.032	.586		.700	-1.585	-4.212	-3.346	-1.673	-6.402
.480	-.465	.474	-.897	.507		.720	-.704	-1.766	-1.376	-1.154	-2.646
.502	-.516	.479	-.846	.454		.750	-.616	-.994	-1.106	-1.199	-1.800
.551	-.471	.489	-.904	.520		.800	-.548	-.569	-.968	-1.115	-1.439
.585	-.426	.497	-.808	.586		.900	-.447	-.229	-.830	-.936	-1.368
.592	-.419	.399	-.827	-.830		.980	-.589	-.288	-.889	-.962	-1.007
.613	-.277	.327	-.699	-.550	Lower	.025	.488	.720	.731	.699	.581
.634	-.258	.229	-.526	-.402		.120	.779	.791	.724	.692	.594
.655	-.219	.098	-.333	-.415		.220	.752	.726	.718	.686	.626
.675	-.136	-.033	-.199	-.283		.300	.657	.687	.672	.628	.529
.696	-.090	.007	-.122	-.092		.620	.718	.785	.731	.628	.155
.774	-.006	.118	-.070	-.072		.750	.853	.890	.738	.705	.581
.852	-.058	.059	-.019	-.053		.850	.616	.713	.560	.551	.465
.930	.019	.007	-.006	-.007		.950	.366	.477	.204	.173	.155

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TABLE 22 (b) Continued Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 50°; δ_f = 47°; $\delta_{a,L}$ = 47°; $\delta_{a,R}$ = 47°; h_s/c = 6.0 h_d/c = 0.0

$C_{\mu,k}$ = 0.010 $C_{\mu,f}$ = 0.012 $C_{\mu,a}$ = 0.004

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
α = 18.8°				
.032	-.191	.650	-.327	.148
.053	-.316	.429	-.441	-.040
.100	-.198	.292	-.534	-.282
.145	-.171	.201	-.487	-.336
.189	-.092	.214	-.434	-.276
.234	-.092	.234	-.120	-.329
.280	-.125	.292	.067	-.343
.326	-.151	.305	[illegible]	-.437
.371	-.342	.396	-.327	-.511
.392	-.422	.458	-.855	-.625
.413	-.501	.520	-1.128	.175
.434	-.547	.559	-1.769	.625
.457	-.514	.550	-1.356	.693
.480	-.514	.541	-1.095	.659
.502	-.514	.532	-1.015	.639
.551	-.395	.514	-.982	.625
.585	-.369	.500	-1.055	.672
.592	-.336	.422	-1.229	-1.197
.613	-.244	.351	-.908	-.600
.634	-.211	.266	-.628	-.403
.655	-.191	.130	-.387	-.390
.675	-.105	.019	-.220	-.249
.696	-.059	.032	-.087	-.074
.774	-.082	.062	-.057	-.050
.852	-.105	.091	-.027	-.027
.930	-.020	.084	-.027	.128
α = 22.9°				
.032	-.224	.767	-.461	.065
.053	-.342	.551	-.561	-.111
.100	-.211	.403	-.621	-.353
.145	-.171	.276	-.574	-.419
.189	-.072	.316	-.521	-.340
.234	-.072	.323	-.140	-.406
.280	-.119	.363	.087	-.451
.326	-.198	.390	-.060	-.569
.371	-.415	.498	-.467	-.641
.392	-.498	.552	-.942	-.844
.413	-.580	.605	-1.262	.190
.434	-.612	.639	-2.057	.680
.457	-.547	.625	-1.563	.726
.480	-.494	.612	-1.349	.680
.502	-.481	.599	-1.149	.661
.551	-.329	.572	-1.095	.621
.585	-.277	.551	-1.149	.648
.592	-.290	.491	-1.549	-1.419
.613	-.184	.403	-1.315	-.700
.634	-.171	.276	-.761	-.523
.655	-.158	.134	-.401	-.327
.675	-.079	[illegible]	-.227	-.105
.696	-.040	.020	-.107	.013
.774	-.040	.067	-.083	.010
.852	-.132	.114	-.060	.007
.930	-.040	.141	-.007	.137

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
α = 18.8°						
Upper	.010	-4.580	-3.015	-3.490	-3.446	-3.300
	.080	-1.728	-1.969	-3.221	-3.312	-1.844
	.130	-3.119	-2.950	-3.167	-3.138	-3.267
	.145	-9.648	-8.784	-8.244	-8.400	-8.838
	.155	-4.145	-4.359	-4.492	-4.367	-4.037
	.180	-2.722	-2.904	-3.174	-3.018	-3.128
	.220	-1.709	-2.079	-2.286	-2.210	-2.239
	.270	-1.286	-1.657	-1.836	-1.736	-1.824
	.400	-.975	-1.247	-1.358	-1.229	-1.633
	.620	-1.027	-1.501	-1.358	-1.135	-1.646
	.685	-3.106	-4.684	-2.286	-3.232	-7.416
	.693	-2.807	-4.866	-3.120	-3.018	-6.606
	.700	-1.819	-3.456	-2.300	-2.270	-5.058
	.720	-.819	-1.507	-1.217	-1.402	-2.239
	.750	-.585	-.858	-1.130	-1.295	-1.600
	.800	-.494	-.494	-1.022	-1.128	-1.390
	.900	-.370	-.247	-.928	-.962	-1.271
	.980	-.455	-.273	-.962	-.928	-.962
Lower	.025	.708	.825	.807	.815	.639
	.120	.871	.819	.780	.748	.566
	.220	.832	.793	.793	.748	.586
	.300	.760	.747	.726	.714	.547
	.620	.760	.793	.746	.668	.145
	.750	.871	.858	.740	.708	.560
	.850	.656	.708	.585	.588	.448
	.950	.448	.487	.215	.220	.198
α = 22.9°						
Upper	.010	-8.849	-4.068	-4.114	-4.006	-3.675
	.080	-2.017	-3.685	-4.160	-3.960	-2.417
	.130	-3.342	-2.925	-3.271	-3.312	-2.404
	.145	-9.783	-8.304	-6.613	-6.263	-6.467
	.155	-4.135	-4.357	-3.938	-3.566	-2.806
	.180	-2.717	-2.985	-2.878	-2.611	-2.200
	.220	-1.701	-2.172	-2.100	-1.956	-1.390
	.270	-1.432	-1.715	-1.668	-1.549	-1.067
	.400	-1.056	-1.271	-1.262	-1.209	-1.014
	.620	-.988	-1.257	-1.092	-1.269	-.948
	.685	-3.900	-3.120	-.850	-1.956	-1.936
	.693	-4.008	-3.557	-1.962	-2.043	-1.561
	.700	-2.616	-2.582	-1.675	-1.442	-1.376
	.720	-1.298	-1.136	-1.001	-1.028	-.994
	.750	-.861	-.740	-.988	-1.008	-1.021
	.800	-.592	-.545	-.935	-.968	-1.014
	.900	-.477	-.451	-.890	-.895	-.948
	.980	-.518	-.370	-.831	-.868	-.883
Lower	.025	.814	.881	.844	.815	.678
	.120	.894	.854	.791	.781	.599
	.220	.867	.841	.811	.795	.672
	.300	.793	820	.772	.768	.593
	.620	.780	820	.759	.714	.263
	.750	.881	881	.805	.688	.619
	.850	.706	740	.608	.568	.481
	.950	.477	471	.242	.220	.191

TABLE 22 Continued (c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
α = -1.7°											
.032	.263	.318	.291	.292	Upper	.010	.894	.857	.819	.822	.790
.053	.049	.080	.051	.031		.080	.367	.300	.242	.215	.257
.100	-.098	-.067	-.108	-.087		.130	-.478	-.710	-.800	-.778	-.735
.145	-.092	-.098	-.082	-.050		.145	-4.569	-4.459	-4.243	-4.738	-4.152
.189	-.024	-.031	-.019	-.012		.155	-1.727	-1.819	-1.799	-1.784	-1.366
.234	-.055	.055	-.038	.006		.180	-1.286	-1.176	-1.303	-1.183	-1.213
.280	-.055	.073	-.038	.031		.220	-.815	-.894	-.937	-.999	-.796
.326	-.049	.092	-.038	.062		.270	-.674	-.802	-.837	-.740	-.631
.371	-.116	.141	-.127	.124		.400	-.723	-.833	-.844	-.746	-.643
.392	-.138	.160	-.019	.347		.620	-1.231	-1.402	-1.210	-.917	-1.047
.413	-.159	.178	-.190	-.037		.685	-6.504	-6.382	-4.777	-2.796	-6.308
.434	-.227	.178	-.418	-.155		.693	-6.535	-6.970	-6.005	-2.986	-5.267
.457	-.251	.198	-.544	-.099		.700	-4.220	-4.985	-4.442	-1.936	-3.803
.480	-.312	.218	-.493	.062		.720	-2.033	-2.156	-1.700	-1.151	-1.439
.502	-.429	.238	-.544	.149		.750	-1.384	-1.323	-1.024	-1.215	-1.041
.551	-.429	.278	-.734	.335		.800	-.943	-.766	-.726	-.892	-.876
.585	-.429	.306	-.911	.267		.900	-.588	-.245	-.639	-.860	-.759
.592	-.404	.269	-1.063	-1.061		.980	.006	.178	-.490	-.797	-.539
.613	-.312	.196	-.873	-.850	Lower	.025	-.147	-.006	.118	.139	.031
.634	-.263	.190	-.645	-.775		.120	-.227	.006	.105	.070	-.006
.655	-.208	.104	-.418	-.174		.220	-.171	-.012	.074	.057	[illegible]
.675	-.129	.031	-.297	-.081		.300	.018	-.049	-.006	[illegible]	-.049
.696	-.061	.018	-.202	-.056		.620	.490	.404	.167	.076	-.043
.774	-.043	-.014	-.076	-.100		.750	.741	.600	.205	.196	.269
.852	-.024	-.037	-.019	-.143		.850	.686	.704	.440	.310	.392
.930	.080	-.227	.095	-.285		.950	.533	.576	.316	.247	.282
α = 5.6°											
.032	.105	.449	.115	.310	Upper	.010	.547	.526	.478	.487	.530
.053	-.118	.205	-.103	.065		.080	-.257	-.269	-.348	-.397	-.288
.100	-.196	.064	-.218	-.116		.130	-1.429	-1.564	-1.781	-1.692	-1.707
.145	-.157	-.026	-.212	-.110		.145	-6.961	-6.397	-6.273	-6.737	-6.384
.189	-.078	.045	-.147	-.045		.155	-2.845	-2.795	-2.904	-2.840	-2.459
.234	-.098	.103	-.051	-.045		.180	-2.068	-1.808	-2.039	-1.878	-2.002
.280	-.098	.128	[illegible]	-.045		.220	-1.324	-1.333	-1.497	-1.487	-1.367
.326	-.111	.128	[illegible]	-.052		.270	-1.001	-1.141	-1.258	-1.147	-1.112
.371	-.235	.212	-.173	-.026		.400	-.883	-.994	-1.136	-.987	-1.033
.392	-.278	.250	-.314	.148		.620	-1.153	-1.462	-1.355	-1.013	-1.439
.413	-.321	.288	-.583	.232		.685	-3.991	-6.276	-4.763	-2.872	-9.236
.434	-.373	.321	-.897	.232		.693	-3.550	-6.827	-6.021	-2.667	-8.314
.457	-.399	.338	-.788	.381		.700	-2.015	-4.833	-4.433	-1.776	-6.306
.480	-.477	.354	-.737	.452		.720	-.935	-2.071	-1.736	-1.218	-2.734
.502	-.563	.371	-.724	.445		.750	-.757	-1.244	-1.097	-1.269	-1.910
.551	-.523	.404	-.769	.478		.800	-.711	-.731	-.916	-1.000	-1.583
.585	-.504	.429	-.846	.529		.900	-.580	-.276	-.826	-.987	-1.361
.592	-.458	.391	-.833	-.839		.980	-.507	-.038	-.703	-.923	-.798
.613	-.347	.282	-.679	-.600	Lower	.025	.178	.410	.374	.385	.235
.634	-.294	.205	-.532	-.555		.120	.244	.359	.348	.333	.052
.655	-.249	.103	-.372	-.478		.220	.566	.442	.516	.410	.373
.675	-.157	-.013	-.231	-.174		.300	.632	.590	.632	.577	.543
.696	-.085	-.006	-.147	-.013		.620	.692	.699	.678	.622	.209
.774	-.026	.090	-.096	.013		.750	.817	.763	.671	.692	.615
.852	-.052	.026	-.045	-.136		.850	.612	.641	.574	.545	.497
.930	.052	-.128	.051	-.148		.950	.382	.494	.219	.179	.235
α = 13.1°											
.032	-.058	.618	-.083	.273	Upper	.010	-.265	-1.535	-1.754	-1.686	-1.375
.053	-.239	.369	-.263	.052		.080	-1.048	-1.012	-1.156	-1.205	-1.078
.100	-.200	.185	-.365	-.175		.130	-2.487	-2.630	-2.995	-2.808	-2.839
.145	-.148	.115	-.314	-.201		.145	-8.700	-8.386	-8.193	-8.654	-8.209
.189	-.084	.146	-.263	-.143		.155	-3.654	-3.891	-4.126	-4.026	-3.478
.234	-.116	.178	-.096	-.169		.180	-2.527	-2.515	-2.826	-2.635	-2.652
.280	-.123	.197	.071	-.182		.220	-1.645	-1.802	-2.027	-1.981	-1.820
.326	-.148	.236	.032	-.247		.270	-1.253	-1.452	-1.670	-1.494	-1.433
.371	-.284	.312	-.199	-.247		.400	-1.061	-1.159	-1.358	-1.173	-1.271
.392	-.333	.376	-.622	-.221		.620	-1.167	-1.522	-1.514	-1.122	-1.691
.413	-.381	.439	-.936	.195		.685	-3.210	-5.871	-4.671	-4.474	-9.673
.434	-.458	.465	-1.250	.520		.693	-2.361	-5.909	-5.477	-2.801	-8.712
.457	-.458	.468	-1.032	.611		.700	-1.466	-4.215	-4.002	-2.038	-6.582
.480	-.478	.471	-.923	.572		.720	-.696	-1.758	-1.631	-1.308	-2.775
.502	-.549	.474	-.840	.513		.750	-.590	-1.006	-1.163	-1.308	-1.897
.551	-.471	.480	-.891	.565		.800	-.537	-.548	-1.040	-1.109	-1.581
.585	-.419	.484	-.885	.624		.900	-.444	-.204	-.955	-1.013	-1.478
.592	-.419	.414	-.859	-.890		.980	-.590	-.267	-.936	-1.064	-1.123
.613	-.297	.325	-.686	-.550	Lower	.025	.471	.720	.702	.699	.574
.634	-.277	.242	-.513	-.409		.120	.789	.790	.715	.686	.568
.655	-.252	.108	-.333	-.429		.220	.743	.726	.728	.686	.600
.675	-.161	-.013	-.205	-.299		.300	.656	.681	.669	.641	.516
.696	-.103	.006	-.109	-.110		.620	.723	.783	.728	.673	.155
.774	-.090	.032	-.057	-.084		.750	.816	.885	.695	.699	.568
.852	-.077	.057	-.006	-.058		.850	.584	.707	.585	.551	.458
.930	.013	.006	.013	[illegible]		.950	.378	.490	.234	.212	.136

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TABLE 22 Concluded
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 50°$; $\delta_f = 47°$; $\delta_{a,L} = 47°$; $\delta_{a,R} = 47°$; $h_s/c = 4.0$ $h_d/c = 0.0$

$C_{\mu,k} = 0.010$ $C_{\mu,f} = 0.012$ $C_{\mu,a} = 0.004$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

$\alpha = 18.8°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.205	.733	-.318	.156
.053	-.348	.517	-.435	-.054
.100	-.218	.307	-.500	-.291
.145	-.205	.203	-.487	-.325
.189	-.123	.242	-.422	-.271
.234	-.116	.288	-.110	-.339
.280	-.136	.288	.091	-.352
.326	-.191	.307	.026	-.447
.371	-.368	.406	-.305	-.521
.392	-.440	.468	-.851	-.616
.413	-.511	.530	-1.143	.169
.434	-.573	.556	-1.754	.623
.457	-.580	.551	-1.390	.704
.480	-.552	.546	-1.143	.657
.502	-.566	.541	-1.033	.609
.551	-.457	.531	-.994	.596
.585	-.430	.523	-1.065	.664
.592	-.382	.510	-1.195	-1.212
.613	-.273	.379	-.890	-.600
.634	-.239	.275	-.585	-.413
.655	-.205	.157	-.351	-.386
.675	-.102	.026	-.175	-.257
.696	-.068	.052	-.084	-.095
.774	-.041	.075	-.058	-.071
.852	-.095	.098	-.032	-.047
.930	-.020	.098	-.006	.108

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-5.275	-3.107	-3.636	-3.469	-3.580
	.080	-1.816	-2.047	-3.359	-3.333	-2.114
	.130	-3.225	-2.970	-3.216	-3.183	-3.519
	.145	-9.969	-8.896	-8.471	-8.439	-9.411
	.155	-4.267	-4.428	-4.632	-4.411	-4.351
	.180	-2.838	-2.943	-3.298	-3.093	-3.396
	.220	-1.769	-2.139	-2.384	-2.293	-2.441
	.270	-1.342	-1.701	-1.930	-1.793	-2.019
	.400	-1.028	-1.302	-1.496	-1.358	-1.800
	.620	-1.082	-1.557	-1.530	-1.208	-1.848
	.685	-3.265	-4.795	-3.101	-3.119	-8.176
	.693	-2.865	-5.010	-4.076	-2.644	-7.276
	.700	-1.769	-3.552	-2.986	-2.027	-5.592
	.720	-.855	-1.524	-1.375	-1.377	-2.510
	.750	-.628	-.863	-1.158	-1.280	-1.855
	.800	-.541	-.451	-1.050	-1.078	-1.623
	.900	-.421	-.183	-1.036	-1.007	-1.459
	.980	-.521	-.242	-1.009	-1.007	-1.118
Lower	.025	.708	.831	.826	.786	.641
	.120	.881	.811	.779	.721	.573
	.220	.821	.765	.792	.721	.614
	.300	.748	.765	.731	.682	.546
	.620	.761	.778	.752	.663	.123
	.750	.875	.877	.718	.682	.559
	.850	.674	.713	.609	.559	.450
	.950	.427	.510	.244	.214	.177

$\alpha = 22.9°$

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.237	.818	-.464	.034
.053	-.362	.634	-.564	-.137
.100	-.211	.406	-.643	-.391
.145	-.191	.307	-.564	-.426
.189	-.079	.327	-.511	-.371
.234	-.079	.353	-.159	-.440
.280	-.132	.366	.066	-.467
.326	-.231	.366	-.060	-.591
.371	-.435	.477	-.471	-.673
.392	-.527	.540	-.975	-.893
.413	-.619	.602	-1.273	.179
.434	-.645	.615	-2.069	.707
.457	-.586	.603	-1.565	.769
.480	-.514	.591	-1.313	.735
.502	-.501	.579	-1.167	.694
.551	-.342	.554	-1.088	.652
.585	-.277	.536	-1.180	.687
.592	-.263	.517	-1.532	-1.415
.613	-.184	.425	-1.306	-.700
.634	-.171	.275	-.789	-.549
.655	-.165	.157	-.431	-.357
.675	-.072	.039	-.232	-.110
.696	-.053	.039	-.119	.000
.774	-.059	.072	-.109	.014
.852	-.158	.105	-.099	.027
.930	-.026	.157	-.046	.165

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-8.804	3.912	-4.382	-4.038	-3.761
	.080	-1.975	3.526	-4.444	-3.979	-2.470
	.130	-3.290	2.832	-3.482	-3.415	-2.410
	.145	-9.609	8.216	-7.005	-6.406	-6.467
	.155	-4.075	4.304	-4.190	-3.660	-2.799
	.180	-2.662	2.950	-3.091	-2.686	-2.239
	.220	-1.675	2.132	-2.253	-1.996	-1.423
	.270	-1.393	1.701	-1.813	-1.631	-1.106
	.400	-1.027	1.262	-1.387	-1.273	-1.054
	.620	-.942	1.289	-1.209	-1.267	-.994
	.685	-3.761	3.284	-.913	-2.049	-1.989
	.693	-3.840	3.794	-2.115	-2.102	-1.614
	.700	-2.486	2.675	-1.779	-1.479	-1.409
	.720	-1.249	1.164	-1.106	-1.088	-1.008
	.750	-.818	-.752	-1.058	-1.074	-1.021
	.800	-.563	-.523	-1.010	-1.028	-1.027
	.900	-.445	-.366	-.968	-.935	-.975
	.980	-.491	-.281	-.900	-.908	-.922
Lower	.025	.818	.857	.859	.829	.652
	.120	.929	.837	.790	.782	.612
	.220	.863	.818	.824	.802	.685
	.300	.805	.811	.776	.769	.586
	.620	.791	.811	.755	.716	.237
	.750	.883	.863	.804	.696	.593
	.850	.693	.720	.618	.570	.461
	.950	.477	.477	.261	.199	.171

TABLE 23 (a)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 00°$; $\delta_f = 00°$; $\delta_{a,L} = 00°$; $\delta_{a,R} = 00°$; $h_s/c = 8.0$ $h_d/c = 4.0$

$C_{\mu,k} = 0.000$ $C_{\mu,f} = 0.000$ $C_{\mu,a} = 0.000$

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

x/l	Fuselage: 0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
$\alpha = -.4°$				
.032	.288	.290	.288	.307
.053	.052	.064	.063	.058
.100	-.086	-.064	-.086	-.087
.145	-.086	-.127	-.081	-.058
.189	-.012	-.052	.000	-.023
.234	-.052	.017	-.058	-.012
.280	-.052	.006	-.069	-.017
.326	-.012	.000	-.046	-.006
.371	-.052	.012	.086	.012
.392	-.063	-.058	.023	.029
.413	-.012	.046	.017	.012
.434	-.017	.075	-.012	-.035
.457	-.017	-.058	-.012	-.070
.480	-.012	-.070	-.023	-.127
.502	-.069	-.046	-.035	-.162
.551	-.035	-.070	-.063	-.145
.585	-.040	-.058	-.052	-.127
.592	-.040	.000	-.052	-.058
.613	-.012	.000	-.035	-.081
.634	-.012	.012	-.012	-.035
.655	-.006	.041	.000	-.006
.675	.006	.000	.035	.012
.696	.035	.023	.040	.029
.774	.000	.093	.023	.023
.852	-.069	-.058	-.040	-.098
.930	-.006	-.006	-.006	-.023
$\alpha = 7.0°$				
.032	.088	.443	.147	.326
.053	-.130	.219	-.088	.076
.100	-.195	.035	-.193	-.111
.145	-.136	-.023	-.182	-.087
.189	-.047	.012	-.094	-.029
.234	-.088	.075	-.070	-.035
.280	-.077	.081	-.018	-.047
.326	-.071	.046	-.059	-.023
.371	-.142	.109	-.135	.023
.392	-.145	.130	-.264	.169
.413	-.147	.144	-.410	.227
.434	-.171	.167	-.381	.181
.457	-.153	.150	-.322	.134
.480	-.142	.130	-.258	.047
.502	-.165	.110	-.193	-.006
.551	-.118	.090	-.147	-.029
.585	-.083	.070	-.123	-.041
.592	-.071	.058	-.106	-.111
.613	-.047	.058	-.070	-.017
.634	-.024	.063	-.064	-.012
.655	-.024	.052	-.018	-.017
.675	.006	.006	.018	.017
.696	.029	.023	.029	.017
.774	-.024	.115	.006	.035
.852	-.094	-.029	-.064	-.093
.930	-.029	.040	-.041	.017
$\alpha = 14.5°$				
.032	-.072	.629	-.083	.265
.053	-.234	.381	-.254	.053
.100	-.192	.175	-.330	-.165
.145	-.132	.079	-.283	-.147
.189	-.048	.121	-.218	-.100
.234	-.066	.187	-.106	-.100
.280	-.042	.206	-.024	-.071
.326	-.030	.181	-.035	-.059
.371	-.108	.230	-.112	.047
.392	-.168	.245	-.171	.265
.413	-.222	.260	-.531	.366
.434	-.313	.266	-.525	.330
.457	-.349	.220	-.543	.265
.480	-.409	.190	-.560	.171
.502	-.481	.150	-.566	.094
.551	-.433	.110	-.619	.000
.585	-.403	.073	-.631	-.029
.592	-.373	.097	-.637	-.045
.613	-.264	.085	-.560	-.065
.634	-.246	.067	-.460	-.059
.655	-.210	.060	-.360	-.088
.675	-.120	.006	-.283	-.071
.696	-.066	.024	-.165	-.071
.774	.000	-.010	-.006	.000
.852	-.066	-.042	-.065	-.071
.930	-.054	.115	-.106	.106

Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
$\alpha = -.4°$						
Upper	.010	.000	.070	.197	.299	.328
	.080	.006	.035	.122	.144	.144
	.130	-.040	-.023	.081	.138	.092
	.145	-.184	-.064	-.023	.138	.092
	.155	-.046	.012	.174	.173	.167
	.180	-.121	-.012	.087	.178	.144
	.220	-.058	-.012	.110	.190	.150
	.270	-.040	-.029	.116	.219	.173
	.400	-.086	-.041	.266	.374	.311
	.620	-.063	.000	.851	.714	.599
	.685					
	.693					
	.700	-.029	-.058	-.359	-.368	-.420
	.720	-.058	-.104	-.371	-.380	-.415
	.750	-.075	-.162	-.382	-.386	-.420
	.800	-.069	-.168	-.382	-.386	-.438
	.900	-.023	-.151	-.423	-.403	-.455
	.980	.006	-.075	-.382	-.397	-.438
Lower	.025	.127	.087	-.012	-.138	-.150
	.120	.029	.006	-.046	-.006	.063
	.220	-.081	-.098	-.180	-.115	-.052
	.300	-.132	-.174	-.185	-.127	-.092
	.620	-.132	-.174	-.788	-.772	-.749
	.750	-.138	-.174	-.417	-.455	-.426
	.850	-.086	-.133	-.104	-.345	-.276
	.950	.006	-.070	-.214	-.294	-.311
$\alpha = 7.0°$						
Upper	.010	-1.826	-.829	-.705	-.727	-.684
	.080	-1.143	-.887	-.758	-.780	-.726
	.130	-.393	-.846	-.758	-.692	-.572
	.145	-.508	-.725	-.647	-.580	-.466
	.155	-.345	-.772	-.711	-.592	-.437
	.180	-.393	-.679	-.641	-.504	-.354
	.220	-.308	-.530	-.536	-.352	-.201
	.270	-.260	-.357	-.408	-.199	-.088
	.400	-.236	-.167	-.117	.041	.083
	.620	-.085	-.023	.227	.299	.336
	.685					
	.693					
	.700	-.073	-.127	-.379	-.340	-.460
	.720	-.115	-.150	-.379	-.346	-.460
	.750	-.127	-.184	-.414	-.369	-.490
	.800	-.097	-.173	-.402	-.387	-.484
	.900	-.036	-.138	-.379	-.393	-.484
	.980	-.048	-.127	-.315	-.352	-.383
Lower	.025	.659	.605	.565	.534	.460
	.120	.351	.328	.280	.311	.295
	.220	.169	.161	.122	.135	.142
	.300	.073	.046	.041	.076	.071
	.620	-.036	-.058	-.699	-.698	-.826
	.750	-.079	-.104	-.192	-.311	-.283
	.850	-.079	-.081	-.052	-.205	-.142
	.950	.000	-.058	-.146	-.205	-.189
$\alpha = 14.5°$						
Upper	.010	-.531	-.550	-.525	-.619	-.637
	.080	-.572	-.538	-.543	-.608	-.631
	.130	-.519	-.562	-.596	-.619	-.637
	.145	-.596	-.562	-.608	-.631	-.631
	.155	-.578	-.544	-.590	-.625	-.625
	.180	-.572	-.550	-.590	-.596	-.619
	.220	-.566	-.575	-.602	-.596	-.625
	.270	-.590	-.611	-.637	-.613	-.643
	.400	-.596	-.659	-.696	-.672	-.691
	.620	-.631	-.653	-.619	-.596	-.619
	.685					
	.693					
	.700	-.596	-.599	-.478	-.619	-.691
	.720	-.596	-.611	-.478	-.619	-.679
	.750	-.555	-.605	-.519	-.643	-.691
	.800	-.490	-.556	-.549	-.643	-.679
	.900	-.372	-.478	-.543	-.613	-.601
	.980	-.378	-.423	-.466	-.543	-.511
Lower	.025	.779	.750	.708	.696	.607
	.120	.484	.472	.431	.442	.397
	.220	.295	.272	.254	.254	.234
	.300	.212	.187	.165	.159	.108
	.620	.000	-.054	-.714	-.861	-.865
	.750	-.088	-.133	-.118	-.218	-.264
	.850	-.124	-.157	-.112	-.183	-.168
	.950	-.153	-.236	-.242	-.301	-.300

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TABLE 23 Continued
(a) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 8.0 h_d/c = 4.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = 20.4°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.150	.761	-.294	.158
.053	-.294	.536	-.415	-.030
.100	-.168	.329	-.475	-.256
.145	-.108	.231	-.403	-.286
.189	-.012	.256	-.337	-.207
.234	.012	.292	-.132	-.237
.280	.036	.310	-.036	-.183
.326	.078	.304	.000	-.134
.371	-.078	.341	-.096	.000
.392	-.165	.348	-.198	.304
.413	-.252	.359	-.523	.456
.434	-.361	.347	-.523	.438
.457	-.403	.300	-.553	.383
.480	-.433	.260	-.601	.262
.502	-.499	.220	-.625	.164
.551	-.547	.180	-.679	.024
.585	-.565	.140	-.685	-.006
.592	-.619	.128	-.709	-.050
.613	-.523	.091	-.721	-.067
.634	-.535	.043	-.661	-.110
.655	-.469	.018	-.511	-.158
.675	-.349	-.030	-.409	-.158
.696	-.264	-.006	-.264	-.176
.774	-.084	-.020	-.108	-.146
.852	-.078	-.037	-.126	-.049
.930	-.108	.176	-.252	.170

Surface	x/c	Wing, flap, or aileron 0.221	0.–26	0.640	0.800	0.918
Upper	.010	-.514	-.469	-.596	-.631	-.625
	.080	-.532	-.463	-.621	-.625	-.619
	.130	-.454	-.481	-.651	-.649	-.619
	.145	-.538	-.475	-.669	-.637	-.619
	.155	-.538	-.475	-.639	-.637	-.613
	.180	-.538	-.463	-.639	-.625	-.613
	.220	-.555	-.469	-.645	-.625	-.619
	.270	-.561	-.487	-.663	-.655	-.613
	.400	-.639	-.536	-.676	-.649	-.637
	.620	-.639	-.621	-.694	-.649	-.625
	.685					
	.693					
	.700	-.687	-.609	-.511	-.613	-.697
	.720	-.675	-.596	-.505	-.625	-.673
	.750	-.687	-.596	-.554	-.625	-.685
	.800	-.669	-.584	-.602	-.649	-.679
	.900	-.573	-.548	-.621	-.685	-.709
	.980	-.508	-.536	-.602	-.679	-.673
Lower	.025	.878	.840	.779	.745	.655
	.120	.585	.584	.529	.517	.445
	.220	.388	.377	.341	.325	.294
	.300	.317	.304	.250	.240	.168
	.620	.042	.006	-.773	-.913	-.559
	.750	-.078	-.116	-.116	-.228	-.258
	.850	-.161	-.176	-.170	-.252	-.252
	.950	-.227	-.323	-.329	-.421	-.415

α = 24.4°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.212	.806	-.429	.072
.053	-.290	.614	-.514	-.108
.100	-.139	.385	-.557	-.317
.145	-.085	.304	-.484	-.340
.189	.024	.316	-.386	-.257
.234	.067	.354	-.165	-.275
.280	.079	.378	-.055	-.221
.326	.018	.366	-.012	-.161
.371	-.139	.409	-.147	.006
.392	-.220	.421	-.159	.364
.413	-.296	.434	-.478	.514
.434	-.363	.391	-.484	.520
.457	-.357	.330	-.484	.448
.480	-.387	.270	-.508	.334
.502	-.478	.210	-.514	.245
.551	-.544	.150	-.576	.096
.585	-.599	.099	-.612	.036
.592	-.653	.124	-.655	-.045
.613	-.562	.081	-.668	-.060
.634	-.587	.068	-.680	-.125
.655	-.575	.025	-.637	-.197
.675	-.448	-.037	-.563	-.257
.696	-.393	-.031	-.447	-.293
.774	-.169	.031	-.233	-.287
.852	-.121	.090	-.208	-.084
.930	-.200	.174	-.484	.185

Surface	x/c	Wing, flap, or aileron 0.221	0.–26	0.640	0.800	0.918
Upper	.010	-.596	-.558	-.472	-.582	-.587
	.080	-.590	-.552	-.484	-.570	-.575
	.130	-.523	-.577	-.514	-.612	-.575
	.145	-.609	-.577	-.555	-.612	-.593
	.155	-.596	-.552	-.532	-.594	-.587
	.180	-.609	-.558	-.532	-.588	-.575
	.220	-.609	-.552	-.520	-.582	-.562
	.270	-.627	-.589	-.555	-.606	-.581
	.400	-.663	-.633	-.603	-.606	-.587
	.620	-.755	-.664	-.603	-.600	-.593
	.685					
	.693					
	.700	-.730	-.670	-.496	-.606	-.611
	.720	-.767	-.664	-.496	-.594	-.587
	.750	-.767	-.664	-.520	-.606	-.593
	.800	-.742	-.639	-.555	-.612	-.599
	.900	-.627	-.602	-.567	-.655	-.623
	.980	-.548	-.608	-.573	-.661	-.647
Lower	.025	.925	.844	.812	.778	.665
	.120	.694	.639	.603	.588	.502
	.220	.511	.434	.424	.398	.357
	.300	.420	.372	.340	.300	.254
	.620	.073	.019	-.669	-.864	-.302
	.750	-.037	-.105	-.084	-.196	-.169
	.850	-.128	-.199	-.125	-.233	-.218
	.950	-.231	-.347	-.317	-.392	-.375

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TABLE 23 (b) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 6.0 h_d/c = 3.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of :

α = -.4°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.256	.277	.330	.292
.053	.041	.041	.065	.040
.100	-.099	-.071	-.106	-.086
.145	-.099	-.130	-.088	-.069
.189	-.017	-.077	-.018	-.029
.234	-.041	.000	-.065	-.023
.280	-.041	.006	-.071	-.029
.326	-.029	-.012	-.018	-.017
.371	-.047	.000	-.053	-.006
.392	-.058	.015	.006	.029
.413	-.017	.029	.000	.017
.434	-.023	.071	-.035	-.023
.457	-.006	.035	-.024	-.057
.480	-.017	-.020	-.035	-.109
.502	-.041	-.071	-.059	-.132
.551	-.035	-.071	-.053	-.120
.585	-.029	-.059	-.047	-.120
.592	-.029	-.006	-.065	-.086
.613	-.017	.029	-.047	-.086
.634	-.006	.029	-.012	-.046
.655	-.012	.024	-.006	-.017
.675	[illegible]	-.006	.018	.017
.696	.047	.018	.035	.023
.774	-.006	.088	.029	.017
.852	-.082	-.065	-.071	-.097
.930	-.012	-.012	-.006	-.017

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	[illegible]	.029	.092	.289	.233
	.080	.006	.012	.046	.112	.099
	.130	-.047	-.053	.017	.083	.012
	.145	-.189	-.088	-.034	.041	.017
	.155	-.053	.006	.120	.088	.099
	.180	-.153	-.041	.040	.106	.070
	.220	-.077	-.053	.040	.112	.076
	.270	-.065	-.059	.052	.112	.070
	.400	-.106	-.077	.132	.218	.169
	.620	-.059	.012	.670	.678	.612
	.685					
	.693					
	.700	-.024	-.053	-.338	-.389	-.437
	.720	-.077	-.088	-.338	-.395	-.449
	.750	-.083	-.124	-.366	-.407	-.460
	.800	-.071	-.136	-.361	-.419	-.460
	.900	-.041	-.118	-.383	-.431	-.472
	.980	-.024	-.053	-.349	-.407	-.425
Lower	.025	.106	.112	.017	-.106	-.087
	.120	.012	.018	-.011	-.006	.087
	.220	-.083	-.088	-.155	-.130	-.064
	.300	-.142	-.183	-.172	-.142	-.099
	.620	-.130	-.147	-.859	-.767	-.682
	.750	-.130	-.159	-.263	-.348	-.297
	.850	-.083	-.106	-.132	-.242	-.192
	.950	-.006	-.047	-.195	-.260	-.274

α = 7.0°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	.106	.475	.155	.321
.053	-.111	.231	-.063	.093
.100	-.188	.047	-.184	-.105
.145	-.129	-.024	-.173	-.082
.189	-.059	.018	-.104	-.029
.234	-.088	.083	-.069	-.035
.280	-.070	.101	-.006	-.041
.326	-.059	.077	-.040	-.029
.371	-.135	.119	-.121	.023
.392	-.155	.140	-.265	.204
.413	-.176	.166	-.426	.245
.434	-.176	.154	-.415	.192
.457	-.152	.125	-.334	.140
.480	-.158	.100	-.265	.058
.502	-.176	.075	-.196	.012
.551	-.123	.045	-.155	-.035
.585	-.094	.036	-.115	-.035
.592	-.082	.071	-.109	-.128
.613	-.053	.059	-.069	-.035
.634	-.035	.059	-.035	.006
.655	-.023	.071	-.023	.006
.675	-.012	.018	.029	.035
.696	.029	.042	.035	.029
.774	-.012	.015	.017	.023
.852	-.088	-.030	-.063	-.093
.930	-.041	.047	-.046	.029

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-1.835	-.932	-.723	-.714	-.715
	.080	-1.180	-.956	-.781	-.783	-.762
	.130	-.431	-.973	-.793	-.760	-.680
	.145	-.531	-.861	-.688	-.668	-.616
	.155	-.395	-.908	-.728	-.691	-.575
	.180	-.442	-.825	-.711	-.639	-.522
	.220	-.342	-.689	-.594	-.507	-.364
	.270	-.289	-.522	-.460	-.374	-.229
	.400	-.265	-.267	-.175	-.092	-.035
	.620	-.100	-.042	.163	.225	.252
	.685					
	.693					
	.700	-.065	-.142	-.455	-.345	-.405
	.720	-.112	-.160	-.460	-.334	-.405
	.750	-.112	-.172	-.490	-.345	-.399
	.800	-.088	-.148	-.449	-.374	-.422
	.900	-.041	-.095	-.321	-.357	-.410
	.980	-.053	-.113	-.245	-.299	-.334
Lower	.025	.655	.629	.559	.553	.457
	.120	.360	.338	.280	.328	.305
	.220	.177	.154	.117	.150	.147
	.300	.094	.053	.047	.081	.053
	.620	-.024	-.053	-.740	-.835	-.803
	.750	-.059	-.083	-.117	-.184	-.111
	.850	-.041	-.047	-.058	-.121	-.100
	.950	.029	-.018	-.099	-.155	-.170

α = 14.5°

Fuselage

x/l	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface
.032	-.074	.633	-.089	.260
.053	-.259	.371	-.255	.030
.100	-.197	.189	-.326	-.181
.145	-.142	.110	-.285	-.163
.189	-.068	.122	-.214	-.103
.234	-.074	.195	-.107	-.115
.280	-.049	.201	-.012	-.097
.326	-.031	.189	-.053	-.067
.371	-.117	.243	-.142	.042
.392	-.157	.247	-.196	.284
.413	-.197	.250	-.522	.357
.434	-.296	.268	-.528	.327
.457	-.357	.220	-.558	.272
.480	-.450	.180	-.594	.157
.502	-.518	.140	-.605	.085
.551	-.493	.100	-.671	-.012
.585	-.462	.061	-.653	-.030
.592	-.438	.116	-.629	-.045
.613	-.302	.067	-.552	-.060
.634	-.296	.067	-.439	-.079
.655	-.222	.055	-.338	-.127
.675	-.136	-.006	-.237	-.079
.696	-.074	.037	-.131	-.085
.774	-.006	[illegible]	-.006	[illegible]
.852	-.055	-.043	-.071	-.091
.930	-.055	.110	-.077	.091

Wing, flap, or aileron

Surface	x/c	0.221	0.426	0.640	0.800	0.918
Upper	.010	-.607	-.536	-.593	-.665	-.678
	.080	-.625	-.511	-.617	-.647	-.678
	.130	-.583	-.517	-.683	-.671	-.703
	.145	-.667	-.517	-.708	-.677	-.696
	.155	-.643	-.511	-.677	-.665	-.690
	.180	-.667	-.499	-.677	-.671	-.696
	.220	-.637	-.529	-.689	-.647	-.678
	.270	-.649	-.566	-.726	-.671	-.703
	.400	-.709	-.609	-.768	-.724	-.764
	.620	-.625	-.657	-.659	-.611	-.641
	.685					
	.693					
	.700	-.577	-.609	-.514	-.617	-.690
	.720	-.583	-.609	-.502	-.605	-.684
	.750	-.535	-.609	-.538	-.641	-.703
	.800	-.469	-.584	-.587	-.653	-.678
	.900	-.313	-.499	-.575	-.629	-.598
	.980	-.306	-.438	-.478	-.522	-.512
Lower	.025	.769	.724	.714	.671	.592
	.120	.481	.463	.429	.415	.382
	.220	.306	.262	.242	.220	.210
	.300	.228	.183	.163	.137	.099
	.620		-.049	-.659	-.760	-.290
	.750	-.078	-.134	-.079	-.142	-.136
	.850	-.096	-.183	-.139	-.202	-.185
	.950	-.114	-.256	-.290	-.315	-.302

TABLE 23 Continued
(b) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 6.0 h_d/c = 3.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

C_p values for spanwise stations, $\frac{y}{b/2}$, of:

α = 10.6°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	.018	.540	.035	.287
.053	-.176	.309	-.153	.064
.100	-.205	.125	-.283	-.141
.145	-.147	.053	-.212	-.129
.189	-.059	.071	-.159	-.064
.234	-.076	.142	-.083	-.076
.280	-.064	.142	-.006	-.070
.326	-.064	.137	-.047	-.064
.371	-.141	.178	-.383	.029
.392	-.180	.200	-.289	.217
.413	-.223	.226	-.613	.287
.434	-.276	.214	-.761	.270
.457	-.287	.180	-.779	.217
.480	-.293	.150	-.631	.111
.502	-.299	.120	-.501	.059
.551	-.205	.090	-.330	.006
.585	-.182	.065	-.236	-.006
.592	-.135	.095	-.189	-.015
.613	-.111	.083	-.136	-.029
.634	-.076	.083	-.100	-.006
.655	-.070	.077	-.065	[illegible]
.675	-.023	.012	-.012	.023
.696	.018	.036	.006	.029
.774	-.006	[illegible]	.029	.023
.852	-.100	-.036	-.071	-.082
.930	-.053	.059	-.065	.059

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.938	.831	-.704	-.761	-.815
	.080	-.950	.843	-.727	-.785	-.850
	.130	-.873	.873	-.792	-.826	-.885
	.145	-.902	.843	-.756	-.814	-.897
	.155	-.902	.873	-.780	-.826	-.909
	.180	-.902	-.867	-.792	-.844	-.909
	.220	-.867	-.861	-.780	-.844	-.909
	.270	-.819	-.825	-.786	-.867	-.874
	.400	-.629	-.712	-.651	-.690	-.510
	.620	-.148	-.196	-.111	.018	.311
	.685					
	.693					
	.700	-.226	-.332	-.393	-.490	-.504
	.720	-.214	-.321	-.405	-.507	-.487
	.750	-.214	-.303	-.405	-.537	-.510
	.800	-.142	-.249	-.352	-.454	-.504
	.900	-.083	-.190	-.246	-.336	-.457
	.980	-.160	-.214	-.246	-.301	-.375
Lower	.025	.742	.700	.663	.667	.563
	.120	.457	.404	.369	.419	.364
	.220	.249	.214	.182	.201	.193
	.300	.166	.125	.135	.118	.082
	.620	.006	-.024	-.698	-.891	-.645
	.750	-.030	-.071	-.064	-.130	-.088
	.850	-.053	-.083	-.064	-.100	-.076
	.950	-.006	-.089	-.106	-.124	-.176

α = 18.4°

x/l	Fuselage 0.000, Upper surface	Fuselage 0.000, Lower surface	Fuselage 0.154, Upper surface	Fuselage 0.154, Lower surface
.032	-.132	.712	-.228	.191
.053	-.288	.510	-.367	-.006
.100	-.162	.297	-.439	-.221
.145	-.102	.190	-.361	-.227
.189	-.024	.237	-.294	-.161
.234	-.018	.261	-.132	-.179
.280	.012	.291	-.024	-.155
.326	.036	.261	-.012	-.113
.371	-.078	.309	-.120	.018
.392	-.018	.324	-.162	.305
.413	-.258	.332	-.505	.412
.434	-.367	.321	-.517	.388
.457	-.403	.270	-.547	.328
.480	-.451	.220	-.553	.245
.502	-.517	.170	-.577	.161
.551	-.535	.120	-.661	.024
.585	-.553	.077	-.721	-.006
.592	-.589	.107	-.727	-.050
.613	-.457	.071	-.715	-.078
.634	-.463	.065	-.625	-.113
.655	-.415	.053	-.499	-.185
.675	-.294	-.024	-.409	-.185
.696	-.222	.000	-.294	-.209
.774	-.042	.030	-.084	-.125
.852	-.066	.100	-.108	-.102
.930	-.078	.178	-.192	.167

Surface	x/c	Wing, flap, or aileron 0.221	0.426	0.640	0.800	0.918
Upper	.010	-.641	-.499	-.484	-.673	-.685
	.080	-.688	-.493	-.526	-.673	-.685
	.130	-.641	-.487	-.579	-.685	-.697
	.145	-.699	-.493	-.603	-.691	-.697
	.155	-.688	-.475	-.573	-.679	-.679
	.180	-.693	-.475	-.573	-.673	-.691
	.220	-.699	-.481	-.591	-.667	-.685
	.270	-.711	-.510	-.633	-.673	-.691
	.400	-.740	-.564	-.693	-.721	-.727
	.620	-.688	-.647	-.699	-.661	-.679
	.685					
	.693					
	.700	-.635	-.623	-.555	-.685	-.727
	.720	-.629	-.617	-.538	-.661	-.703
	.750	-.589	-.635	-.567	-.691	-.703
	.800	-.478	-.599	-.603	-.703	-.709
	.900	-.274	-.546	-.639	-.727	-.703
	.980	-.321	-.499	-.597	-.673	-.673
Lower	.025	.851	.783	.741	.721	.643
	.120	.571	.522	.478	.481	.427
	.220	.379	.326	.287	.270	.252
	.300	.291	.255	.209	.180	.144
	.620	.041	-.018	-.597	-.577	-.246
	.750	-.058	-.113	-.102	-.156	-.168
	.850	-.105	-.178	-.197	-.264	-.222
	.950	-.134	-.297	-.346	-.415	-.391

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TABLE 23 Continued (c)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 4.0 h_d/c = 2.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

x/l	Fuselage: 0.000, Upper surface	Fuselage: 0.000, Lower surface	Fuselage: 0.154, Upper surface	Fuselage: 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron: 0.221	0.426	0.640	0.800	0.918
							C_p values for spanwise stations, $\frac{y}{b/2}$, of:				
α = -.5°											
.032	.266	.276	.276	.276	Upper	.010	-.012	.040	.012	.106	.185
.053	.041	.058	.053	.041		.080	.000	.000	-.018	.000	.052
.100	-.116	-.081	-.123	-.123		.130	-.047	-.058	-.053	-.023	-.029
.145	-.104	-.121	-.100	-.059		.145	-.205	-.086	-.094	-.059	-.035
.189	-.035	-.058	-.035	-.041		.155	-.047	-.017	-.006	.000	.058
.234	-.058	-.006	-.059	-.035		.180	-.135	-.063	-.035	-.012	-.006
.280	-.064	.017	-.076	-.012		.220	-.070	-.058	-.041	-.012	-.006
.326	-.017	-.017	-.082	-.018		.270	-.064	-.086	-.041	-.012	-.012
.371	-.058	.000	-.100	-.012		.400	-.123	-.109	-.012	.023	.017
.392	-.064	.010	-.012	.023		.620	-.064	-.035	.322	.340	.336
.413	-.035	.040	-.035	.035		.685					
.434	-.046	.052	-.076	-.029		.693					
.457	-.029	.000	-.018	-.053		.700	-.012	-.046	-.481	-.428	-.434
.480	-.041	-.058	-.059	-.123		.720	-.059	-.086	-.457	-.410	-.429
.502	-.070	-.063	-.082	-.141		.750	-.070	-.092	-.475	-.405	-.429
.551	-.041	-.075	-.082	-.135		.800	-.047	-.104	-.457	-.369	-.429
.585	-.035	-.069	-.076	-.117		.900	-.006	-.063	-.369	-.340	-.394
.592	-.035	-.012	-.070	-.111		.980	.006	-.023	-.199	-.317	-.284
.613	-.035	-.006	-.064	-.070	Lower	.025	.135	.098	.035	.029	-.058
.634	-.017	.006	-.029	-.029		.120	.035	.017	-.018	.035	.093
.655	-.017	.023	.000	-.012		.220	-.070	-.069	-.164	-.100	-.075
.675	-.017	-.006	.018	.018		.300	-.135	-.173	-.164	-.141	-.104
.696	.023	.023	.018	.018		.620	-.100	-.132	-.528	-.592	-.684
.774	-.017	.092	.006	.029		.750	-.111	-.109	-.164	-.176	-.145
.852	-.081	-.075	-.059	-.106		.850	-.053	-.069	-.100	-.158	-.127
.930	-.035	-.006	-.018	-.029		.950	.023	-.012	-.100	-.176	-.156
α = 6.9°											
.032	.094	.463	.129	.307	Upper	.010	-1.935	-.973	-.802	-.797	-.802
.053	-.106	.214	-.070	.083		.080	-1.372	-1.021	-.879	-.862	-.873
.100	-.201	.053	-.188	-.112		.130	-.446	-1.033	-.891	-.844	-.785
.145	-.147	-.012	-.170	-.088		.145	-.498	-.914	-.796	-.762	-.708
.189	-.059	.006	-.106	-.047		.155	-.364	-.973	-.838	-.792	-.678
.234	-.094	.065	-.070	-.047		.180	-.410	-.902	-.820	-.727	-.590
.280	-.083	.077	-.023	-.047		.220	-.328	-.742	-.708	-.586	-.431
.326	-.065	.053	-.059	-.035		.270	-.281	-.505	-.590	-.457	-.301
.371	-.142	.107	-.158	.012		.400	-.258	-.255	-.283	-.193	-.118
.392	.006	.135	-.264	.153		.620	-.117	-.089	.024	.070	.142
.413	-.165	.160	-.422	.236		.685					
.434	-.195	.166	-.399	.195		.693					
.457	-.165	.140	-.317	.147		.700	-.053	-.101	-.496	-.381	-.372
.480	-.159	.115	-.252	.071		.720	-.100	-.107	-.496	-.375	-.354
.502	-.159	.090	-.211	.018		.750	-.088	-.107	-.425	-.399	-.366
.551	-.130	.060	-.152	-.006		.800	-.088	-.101	-.295	-.352	-.366
.585	-.088	.036	-.129	-.024		.900	-.035	-.047	-.142	-.252	-.295
.592	-.071	.065	-.111	-.118		.980	-.018	-.024	-.083	-.193	-.212
.613	-.035	.065	-.070	-.018	Lower	.025	.668	.623	.572	.557	.478
.634	-.018	.065	-.035	-.012		.120	.340	.332	.271	.311	.313
.655	-.024	.059	-.006	.006		.220	.170	.172	.106	.135	.142
.675	.018	.012	.012	.035		.300	.082	.071	.041	.053	.053
.696	.035	.042	.023	.041		.620	-.018	-.042	-.248	-.387	-.366
.774	-.006	.119	-.012	.035		.750	-.047	-.071	-.059	-.070	-.047
.852	-.083	-.030	-.094	-.083		.850	-.047	-.024	-.047	-.082	-.083
.930	-.024	.036	-.035	.024		.950	.023	.012	-.035	-.100	-.130
α = 10.6°											
.032		.537	.042	.302	Upper	.010	-1.302	-.920	-.768	-.730	-.846
.053	-.207	.307	-.154	.079		.080	-1.290	-.920	-.792	-.766	-.876
.100	-.219	.130	-.279	-.139		.130	-1.079	-.956	-.853	-.801	-.913
.145	-.262	.059	-.231	-.133		.145	-1.085	-.914	-.810	-.795	-.913
.189	-.079	.071	-.148	-.060		.155	-1.067	-.944	-.822	-.813	-.895
.234	-.110	.130	-.071	-.073		.180	-1.014	-.938	-.847	-.807	-.919
.280	-.079	.136	-.012	-.060		.220	-.926	-.932	-.847	-.819	-.913
.326	-.085	.130	-.030	-.054		.270	-.797	-.908	-.847	-.849	-.925
.371	-.170	.183	-.160	.012		.400	-.457	-.726	-.738	-.760	-.761
.392	-.215	.200	-.309	.224		.620	-.170	-.260	-.351	-.326	-.189
.413	-.262	.224	-.665	.321		.685					
.434	-.310	.218	-.861	.272		.693					
.457	-.292	.180	-.861	.224		.700	-.141	-.265	-.308	-.457	-.639
.480	-.298	.150	-.677	.151		.720	-.106	-.248	-.321	-.451	-.694
.502	-.329	.120	-.516	.073		.750	-.106	-.224	-.308	-.398	-.651
.551	-.201	.090	-.309	.018		.800	-.070	-.171	-.272	-.303	-.493
.585	-.164	.083	-.196	.012		.900	-.041	-.106	-.181	-.214	-.268
.592	-.158	.094	-.172	.		.980	-.023	-.094	-.145	-.148	-.158
.613	-.103	.094	-.107	-.012	Lower	.025	.774	.708	.677	.671	.554
.634	-.073	.100	-.065	.012		.120	.463	.413	.399	.410	.359
.655	-.055	.083	-.042	.012		.220	.264	.212	.200	.220	.189
.675	-.018	.035	.006	.036		.300	.182	.142	.127	.137	.079
.696	.018	.047	.036	.030		.620	.012	-.012	-.175	-.178	-.091
.774	-.018	.025	.030	.036		.750	-.035	-.053	-.024	-.030	-.061
.852	-.116	.	-.083	.050		.850	-.029	-.047	-.067	-.059	-.079
.930	-.055	.083	-.053	.060		.950	.	-.053	-.073	-.083	-.122

TABLE 23 Continued
(c) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 4.0 h_d/c = 2.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

	C_p values for spanwise stations, $\frac{y}{b/2}$, of :										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.436	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 14.5°											
.032	-.084	.627	-.090	.249	Upper	.010	-.694	-.645	-.611	-.675	-.717
.053	-.233	.394	-.239	.047		.080	-.712	-.633	-.629	-.681	-.711
.100	-.191	.191	-.340	-.154		.130	-.683	-.645	-.671	-.687	-.717
.145	-.119	.102	-.275	-.154		.145	-.766	-.651	-.677	-.699	-.711
.189	-.054	.119	-.215	-.113		.155	-.742	-.651	-.665	-.699	-.705
.234	-.066	.179	-.090	-.113		.180	-.754	-.645	-.665	-.693	-.711
.280	-.048	.185	-.006	-.113		.220	-.736	-.651	-.677	-.693	-.705
.326	-.018	.185	-.024	-.077		.270	-.783	-.693	-.718	-.711	-.735
.371	-.108	.221	-.102	.018		.400	-.742	-.770	-.766	-.770	-.776
.392	-.160	.240	-.221	.237		.620	-.599	-.681	-.742	-.687	-.669
.413	-.215	.263	-.609	.344		.685					
.434	-.311	.233	-.627	.321		.693					
.457	-.376	.200	-.669	.261		.700	-.540	-.603	-.469	-.579	-.621
.480	-.436	.165	-.675	.178		.720	-.499	-.591	-.451	-.555	-.609
.502	-.502	.130	-.669	.101		.750	-.451	-.551	-.475	-.579	-.597
.551	-.442	.095	-.645	[illegible]		.800	-.386	-.538	-.493	-.555	-.543
.585	-.376	.066	-.621	-.036		.900	-.231	-.430	-.499	-.549	-.514
.592	-.370	.119	-.591	-.045		.980	-.261	-.354	-.433	-.466	-.442
.613	-.269	.084	-.508	-.053	Lower	.025	.807	.741	.712	.705	.603
.634	-.233	.072	-.364	-.047		.120	.510	.442	.410	.466	.382
.655	-.185	.066	-.221	-.071		.220	.321	.233	.220	.233	.209
.675	-.108	.006	-.137	-.047		.300	.237	.185	.160	.155	.090
.696	-.042	.042	-.042	-.047		.620	.042	-.018	-.095	-.125	-.125
.774	-.006	.010	.018	[illegible]		.750	-.024	-.096	-.071	-.084	-.113
.852	-.066	-.012	-.054	-.077		.850	-.077	-.119	-.154	-.161	-.185
.930	-.066	.119	-.066	.095		.950	-.077	-.203	-.255	-.275	-.269
α = 18.4°											
.032	-.125	.697	-.210	.204	Upper	.010	-.700	-.6[illegible]1	-.649	-.679	-.689
.053	-.267	.481	-.373	.012		.080	-.706	-.5[illegible]5	-.661	-.697	-.694
.100	-.166	.276	-.433	-.210		.130	-.676	-.5[illegible]3	-.703	-.715	-.700
.145	-.113	.192	-.361	-.210		.145	-.761	-.5[illegible]1	-.697	-.727	-.706
.189	-.012	.210	-.306	-.156		.155	-.755	-.5[illegible]1	-.685	-.715	-.700
.234	-.006	.258	-.138	-.180		.180	-.773	-.5[illegible]3	-.685	-.721	-.694
.280	.012	.276	-.024	-.144		.220	-.755	-.5[illegible]1	-.679	-.709	-.694
.326	.036	.264	.006	-.108		.270	-.791	-.5[illegible]7	-.703	-.727	-.718
.371	-.059	.306	-.084	[illegible]		.400	-.779	-.6[illegible]1	-.745	-.739	-.742
.392	-.006	.320	-.222	.294		.620	-.761	-.6[illegible]7	-.799	-.733	-.712
.413	-.231	.331	-.619	.427		.685					
.434	-.344	.325	-.625	.397		.693					
.457	-.374	.290	-.655	.337		.700	-.645	-.6[illegible]3	-.457	-.601	-.665
.480	-.410	.255	-.685	.228		.720	-.645	-.6[illegible]9	-.445	-.589	-.641
.502	-.475	.220	-.715	.162		.750	-.602	-.6[illegible]3	-.487	-.613	-.641
.551	-.522	.185	-.733	.042		.800	-.469	-.6[illegible]7	-.553	-.625	-.641
.585	-.528	.150	-.739	.012		.900	-.262	-.5[illegible]3	-.565	-.661	-.635
.592	-.552	.132	-.757	-.028		.980	-.365	-.4[illegible]9	-.541	-.625	-.623
.613	-.457	.090	-.667	-.048	Lower	.025	.882	.8[illegible]1	.763	.757	.635
.634	-.451	.066	-.517	-.060		.120	.584	.5[illegible]3	.499	.517	.421
.655	-.404	.066	-.367	-.084		.220	.389	.3[illegible]5	.294	.306	.243
.675	-.303	-.012	-.258	-.060		.300	.298	.2[illegible]4	.210	.216	.142
.696	-.226	.036	-.138	-.084		.620	.030	[illegible]	-.060	-.078	-.131
.774	-.006	.174	-.024	-.030		.750	-.067	-.1[illegible]8	-.108	-.120	-.148
.852	-.059	.050	-.072	-.024		.850	-.116	-.1[illegible]2	-.180	-.210	-.231
.930	-.077	.180	-.144	.156		.950	-.134	-.2[illegible]2	-.294	-.349	-.374

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TABLE 23 Continued
(d)

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

$\delta_n = 00°$; $\delta_f = 00°$; $\delta_{a,L} = 00°$; $\delta_{a,R} = 00°$; $h_s/c = 2.0$ $h_d/c = 1.0$

$C_{\mu,k} = 0.000$ $C_{\mu,f} = 0.000$ $C_{\mu,a} = 0.000$

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = -.5°											
.032	.246	.299	.281	.288	Upper	.010	-.017	-.017	.	.147	.155
.053	.023	.069	.053	.058		.080	.	.	-.012	.006	.011
.100	-.114	-.063	-.100	-.075		.130	-.058	-.081	-.052	-.047	-.069
.145	-.114	-.115	-.082	-.081		.145	-.209	-.104	-.109	-.059	-.069
.189	-.023	-.063	.	-.029		.155	-.041	-.029	-.012	-.012	.
.234	-.069	.	-.047	-.017		.180	-.151	-.086	-.040	-.023	-.046
.280	-.057	.023	-.064	-.017		.220	-.087	-.092	-.069	-.047	-.046
.326	-.034	-.017	-.041	-.017		.270	-.070	-.092	-.092	-.047	-.063
.371	-.063	.012	-.053	-.006		.400	-.110	-.138	-.104	-.070	-.080
.392	-.057	-.052	.006	.029		.620	-.110	-.092	.029	.047	.069
.413	-.046	.058	.012	.029		.685					
.434	-.057	.081	-.029	-.012		.693					
.457	-.034	-.069	-.023	-.058		.700	.012	-.035	-.397	-.387	-.361
.480	-.052	-.058	-.053	-.104		.720	-.052	-.063	-.357	-.346	-.321
.502	-.080	-.052	-.059	-.127		.750	-.064	-.081	-.311	-.305	-.315
.551	-.057	-.046	-.064	-.121		.800	-.058	-.063	-.178	-.223	-.269
.585	-.040	-.035	-.053	-.092		.900	-.012	-.040	-.023	-.059	-.092
.592	-.057	.012	-.047	-.098		.980	.041	.029	.058	.023	.006
.613	-.034	.017	-.023	-.058							
.634	-.023	.023	.006	-.012	Lower	.025	.145	.127	.092	-.006	-.052
.655	-.023	.040	.018	.017		.120	.035	.035	.	.029	.086
.675	-.011	.	.041	.040		.220	-.070	-.063	-.132	-.100	-.069
.696	.017	.023	.035	.023		.300	-.127	-.161	-.167	-.135	-.109
.774	-.011	.104	.018	.035		.620	-.087	-.104	-.121	-.088	-.097
.852	-.015	-.069	-.035	.020		.750	-.098	-.092	-.104	-.106	-.103
.930	-.023	.017	-.006	-.012		.850	-.058	-.052	-.069	-.059	-.057
						.950	.029	.017	.017	-.006	-.017
α = 6.9°											
.032	.098	.478	.152	.324	Upper	.010	-2.080	-1.037	-.814	-.809	-.805
.053	-.098	.227	-.064	.088		.080	-1.183	-1.055	-.867	-.868	-.869
.100	-.168	.052	-.193	-.100		.130	-.373	-1.066	-.903	-.874	-.846
.145	-.133	-.023	-.158	-.100		.145	-.501	-.950	-.785	-.786	-.770
.189	-.058	.012	-.094	-.041		.155	-.338	-1.008	-.844	-.838	-.759
.234	-.093	.082	-.070	-.053		.180	-.414	-.938	-.826	-.768	-.695
.280	-.070	.093	-.006	-.035		.220	-.315	-.793	-.749	-.645	-.539
.326	-.064	.087	-.053	-.035		.270	-.274	-.554	-.625	-.510	-.377
.371	-.156	.122	-.147	.006		.400	-.251	-.268	-.324	-.276	-.220
.392	-.160	.144	-.276	.177		.620	-.152	-.128	-.106	-.106	-.075
.413	-.168	.169	-.428	.236		.685					
.434	-.185	.175	-.393	.195		.693					
.457	-.156	.150	-.305	.147		.700	-.035	-.076	-.254	-.252	-.371
.480	-.156	.125	-.258	.065		.720	-.082	-.093	-.201	-.223	-.336
.502	-.180	.100	-.199	.012		.750	-.082	-.087	-.147	-.188	-.272
.551	-.098	.075	-.152	-.018		.800	-.076	-.064	-.088	-.141	-.162
.585	-.081	.041	-.106	-.018		.900	-.006	-.035	-.053	-.076	-.075
.592	-.075	.070	-.094	-.112		.980	.029	.012	-.035	-.035	-.017
.613	-.035	.064	-.064	-.018							
.634	-.006	.064	-.035	.006	Lower	.025	.670	.647	.584	.563	.504
.655	-.006	.064	-.006	.012		.120	.373	.338	.301	.311	.307
.675	.029	.017	.023	.035		.220	.192	.169	.112	.158	.151
.696	.058	.047	.029	.035		.300	.099	.087	.041	.088	.064
.774	.	.122	.018	.024		.620	-.006	-.023	-.035	.	-.046
.852	-.087	-.017	-.070	-.094		.750	-.029	-.047	-.083	-.059	-.064
.930	-.035	.041	-.035	.018		.850	-.029	-.029	-.035	-.070	-.058
						.950	.047	.017	-.012	-.047	-.046
α = 10.6°											
.032	.006	.537	.035	.299	Upper	.010	-1.340	-1.032	-.768	-.718	-.752
.053	-.175	.313	-.145	.082		.080	-1.316	-1.021	-.803	-.741	-.793
.100	-.192	.118	-.261	-.129		.130	-1.112	-1.050	-.856	-.765	-.810
.145	-.146	.041	-.209	-.117		.145	-1.118	-1.009	-.815	-.770	-.822
.189	-.058	.083	-.156	-.059		.155	-1.130	-1.038	-.827	-.770	-.816
.234	-.082	.136	-.070	-.082		.180	-1.076	-1.038	-.833	-.788	-.816
.280	-.064	.147	.	-.059		.220	-.992	-1.021	-.850	-.805	-.822
.326	-.058	.130	-.041	-.064		.270	-.883	-1.003	-.844	-.823	-.828
.371	-.157	.195	-.162	.012		.400	-.529	-.708	-.762	-.770	-.723
.392	-.200	.205	-.336	.223		.620	-.258	-.324	-.416	-.446	-.385
.413	-.245	.218	-.689	.305		.685					
.434	-.286	.212	-.869	.270		.693					
.457	-.280	.185	-.857	.223		.700	-.144	-.189	-.270	-.307	-.431
.480	-.291	.160	-.614	.141		.720	-.144	-.165	-.258	-.272	-.350
.502	-.303	.135	-.469	.088		.750	-.132	-.165	-.240	-.284	-.291
.551	-.192	.110	-.266	.		.800	-.096	-.124	-.223	-.255	-.239
.585	-.163	.083	-.209	.006		.900	-.048	-.071	-.164	-.209	-.221
.592	-.140	.118	-.162	.003		.980	-.042	-.053	-.123	-.162	-.152
.613	-.087	.100	-.110	.							
.634	-.052	.100	-.058	.	Lower	.025	.769	.714	.668	.666	.554
.655	-.052	.088	-.035	.018		.120	.457	.419	.387	.394	.350
.675	-.012	.018	-.006	.035		.220	.246	.254	.211	.209	.186
.696	.023	.047	.017	.023		.300	.192	.159	.129	.133	.093
.774	-.006	.010	.	.041		.620	.012	.006	-.012	-.006	-.087
.852	-.111	-.006	-.025	.050		.750	-.024	-.024	-.059	-.081	-.093
.930	-.052	.065	-.058	.059		.850	-.030	-.024	-.053	-.104	-.093
						.950	.006	-.018	-.059	-.098	-.128

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TABLE 23 Continued
(d) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 2.0 h_d/c = 1.0

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = 14.4° 1											
.032	-.060	.647	-.097	.263	Upper	.010	-.608	-.6[illegible]9	-.609	-.671	-.729
.053	-.239	.425	-.266	.060		.080	-.626	-.6[illegible]6	-.651	-.695	-.747
.100	-.167	.192	-.339	-.179		.130	-.597	-.6[illegible]8	-.681	-.714	-.741
.145	-.125	.093	-.302	-.173		.145	-.637	-.670	-.687	-.714	-.753
.189	-.048	.140	-.230	-.108		.155	-.626	-.670	-.681	-.708	-.764
.234	-.054	.198	-.121	-.119		.180	-.626	-.682	-.675	-.714	-.764
.280	-.042	.210	-.030	-.096		.220	-.649	-.693	-.705	-.726	-.758
.326	-.030	.192	-.024	-.060		.270	-.637	-.699	-.729	-.744	-.782
.371	-.119	.239	-.121	.084		.400	-.678	-.746	-.794	-.786	-.800
.392	-.175	.254	-.242	.263		.620	-.602	-.693	-.764	-.738	-.681
.413	-.233	.268	-.659	.382		.685					
.434	-.323	.268	-.665	.340		.693					
.457	-.382	.225	-.714	.281		.700	-.556	-.5[illegible]1	-.520	-.575	-.633
.480	-.424	.185	-.738	.191		.720	-.539	-.5[illegible]1	-.520	-.575	-.597
.502	-.514	.145	-.744	.119		.750	-.481	-.5[illegible]8	-.579	-.581	-.579
.551	-.424	.105	-.720	.006		.800	-.417	-.4[illegible]8	-.561	-.593	-.573
.585	-.370	.076	-.665	-.012		.900	-.272	-.3[illegible]7	-.543	-.593	-.555
.592	-.352	.117	-.605	-.075		.980	-.278	-.3[illegible]6	-.448	-.514	-.508
.613	-.251	.082	-.496	-.060							
.634	-.209	.093	-.369	-.042	Lower	.025	.794	.7[illegible]8	.717	.689	.615
.655	-.167	.087	-.230	-.060		.120	.504	.478	.430	.454	.382
.675	-.108	.035	-.121	-.054		.220	.324	.274	.239	.248	.233
.696	-.048	.064	-.054	-.060		.300	.226	.192	.173	.163	.108
.774	-.012	.152	.012	.012		.620	.029	[illegible]	-.048	-.030	-.125
.852	-.108	-.012	-.054	-.066		.750	-.058	-.070	-.113	-.145	-.167
.930	-.060	.122	-.073	.108		.850	-.075	-.093	-.155	-.194	-.209
						.950	-.098	-.163	-.245	-.266	-.311
α = 18.4° 14											
.032	-.138	.744	-.197	.197	Upper	.010	-.689	-.6[illegible]1	-.502	-.597	-.661
.053	-.246	.532	-.346	.024		.080	-.708	-.6[illegible]1	-.526	-.615	-.661
.100	-.156	.302	-.400	-.215		.130	-.701	-.6[illegible]9	-.561	-.615	-.667
.145	-.108	.194	-.340	-.215		.145	-.774	-.6[illegible]3	-.573	-.621	-.661
.189	-.036	.218	-.269	-.149		.155	-.750	-.6[illegible]7	-.573	-.615	-.667
.234	-.018	.266	-.113	-.179		.180	-.774	-.5[illegible]7	-.561	-.603	-.661
.280	.012	.290	-.018	-.143		.220	-.780	-.6[illegible]5	-.585	-.639	-.655
.326	.036	.272	.	-.113		.270	-.804	-.6[illegible]9	-.609	-.639	-.685
.371	-.072	.314	-.084	.024		.400	-.798	-.6[illegible]7	-.681	-.675	-.697
.392	.335	-.006	-.167	.299		.620	-.756	-.7[illegible]0	-.788	-.699	-.715
.413	-.228	.357	-.496	.430		.685					
.434	-.331	.321	-.508	.394		.693					
.457	-.385	.270	-.508	.334		.700	-.714	-.7[illegible]1	-.538	-.591	-.709
.480	-.433	.220	-.555	.227		.720	-.708	-.6[illegible]9	-.532	-.603	-.643
.502	-.487	.170	-.591	.143		.750	-.641	-.6[illegible]3	-.591	-.627	-.649
.551	-.523	.120	-.645	.024		.800	-.520	-.6[illegible]5	-.639	-.669	-.649
.585	-.559	.079	-.705	-.024		.900	-.339	-.5[illegible]6	-.645	-.669	-.667
.592	-.577	.133	-.747	.055		.980	-.369	-.4[illegible]4	-.573	-.615	-.649
.613	-.469	.097	-.729	-.090							
.634	-.457	.073	-.615	-.108	Lower	.025	.859	.8[illegible]5	.764	.729	.661
.655	-.409	.054	-.520	-.119		.120	.593	.5[illegible]6	.484	.472	.451
.675	-.282	-.012	-.424	-.149		.220	.387	.3[illegible]5	.281	.275	.270
.696	-.216	.030	-.317	-.179		.300	.302	.2[illegible]0	.209	.209	.162
.774	-.012	.169	-.102	-.096		.620	.024	.0[illegible]2	-.060	-.030	-.132
.852	-.066	.175	-.119	-.048		.750	-.054	-.0[illegible]7	-.155	-.149	-.192
.930	-.084	.181	-.185	.149		.850	-.121	-.1[illegible]9	-.233	-.245	-.252
						.950	-.121	-.2[illegible]0	-.334	-.382	-.409

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TABLE 23 (e) Continued

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 00°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 1.0 h_d/c = 0.5

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

	C_p values for spanwise stations, $\frac{y}{b/2}$, of:										
	0.000, Upper surface	0.000, Lower surface	0.154, Upper surface	0.154, Lower surface			0.221	0.426	0.640	0.800	0.918
x/l	Fuselage				Surface	x/c	Wing, flap, or aileron				
α = −.5°											
.032	.264	.288	.269	.284		.010	.006	.023	−.012	.206	.217
.053	.035	.057	.052	.046		.080	.011	[illegible]	−.023	.040	.035
.100	−.100	−.074	−.109	−.093		.130	−.029	−.062	−.070	−.006	−.053
.145	−.100	−.113	−.080	−.075		.145	−.172	−.090	−.127	−.046	−.059
.189	−.018	−.062	−.006	−.023		.155	−.052	−.023	−.041	.011	.012
.234	−.053	.006	−.029	−.017		.180	−.137	−.062	−.070	.006	−.023
.280	−.035	.006	−.069	−.012		.220	−.074	−.079	−.087	−.034	−.041
.326	−.006	−.011	−.034	−.006		.270	−.063	−.090	−.098	−.034	−.076
.371	−.035	.006	−.046	[illegible]	Upper	.400	−.120	−.124	−.139	−.074	−.106
.392	−.059	−.068	.011	.046		.620	−.120	−.096	−.087	−.046	−.041
.413	−.023	.040	−.011	.046		.685					
.434	−.023	.057	−.057	−.006		.693					
.457	−.012	−.074	−.034	−.046		.700	−.006	−.028	−.232	−.217	−.217
.480	−.029	−.068	−.057	−.104		.720	−.052	−.057	−.209	−.206	−.188
.502	−.064	−.057	−.057	−.116		.750	−.063	−.051	−.151	−.160	−.182
.551	−.035	−.068	−.057	−.116		.800	−.052	−.045	−.075	−.069	−.094
.585	−.029	−.045	−.057	−.081		.900	−.011	−.023	−.023	−.023	−.035
.592	−.029	[illegible]	−.057	−.087		.980	.057	.045	.058	.040	.041
.613	[illegible]	.006	−.023	−.046							
.634	.012	.017	−.011	−.012		.025	.126	.113	.070	−.057	−.064
.655	.006	.028	.011	[illegible]		.120	.040	.034	.012	.011	.076
.675	.018	−.006	.023	.035		.220	−.074	−.057	−.133	−.103	−.064
.696	.053	.017	.034	.058	Lower	.300	−.120	−.147	−.145	−.132	−.094
.774	−.006	.074	.017	.023		.620	−.092	−.096	−.139	−.086	−.076
.852	−.100	−.085	−.052	−.093		.750	−.103	−.096	−.122	−.097	−.094
.930	−.012	[illegible]	[illegible]	−.012		.850	−.052	−.045	−.064	−.069	−.059
						.950	.017	.034	[illegible]	.006	.006
α = 6.9°											
.032	.071	.472	.147	.307		.010	−2.090	−1.056	−.838	−.833	−.855
.053	−.136	.224	−.070	.071		.080	−.985	−1.091	−.867	−.891	−.914
.100	−.206	.053	−.193	−.118		.130	−.334	−1.115	−.920	−.891	−.879
.145	−.159	−.012	−.170	−.106		.145	−.530	−.985	−.808	−.821	−.808
.189	−.071	.029	−.094	−.047		.155	−.340	−1.056	−.861	−.856	−.785
.234	−.100	.077	−.064	−.041		.180	−.443	−.973	−.849	−.797	−.714
.280	−.083	.088	−.006	−.047		.220	−.334	−.796	−.755	−.686	−.560
.326	−.077	.083	−.053	−.047		.270	−.282	−.590	−.649	−.580	−.425
.371	−.147	.112	−.135	.006	Upper	.400	−.259	−.271	−.342	−.317	−.254
.392	−.158	.145	−.276	.159		.620	−.184	−.165	−.165	−.158	−.171
.413	−.171	.171	−.434	.230		.685					
.434	−.206	.183	−.416	.189		.693					
.457	−.183	.155	−.334	.130		.700	−.046	−.077	−.159	−.211	−.265
.480	−.171	.130	−.276	.053		.720	−.098	−.094	−.147	−.193	−.236
.502	−.177	.105	−.217	.006		.750	−.104	−.088	−.118	−.152	−.189
.551	−.130	.080	−.152	−.035		.800	−.075	−.100	−.094	−.123	−.136
.585	−.088	.059	−.135	−.029		.900	−.029	−.041	−.071	−.088	−.112
.592	−.083	.088	−.094	−.100		.980	.029	.018	[illegible]	−.035	−.029
.613	−.047	.071	−.076	−.018							
.634	−.029	.083	−.029	[illegible]		.025	.668	.643	.602	.569	.490
.655	−.035	.077	−.006	.018		.120	.351	.342	.313	.322	.289
.675	−.012	.024	.023	.047		.220	.173	.171	.118	.141	.147
.696	.029	.065	.041	.041	Lower	.300	.092	.077	.047	.076	.053
.774	−.029	.030	.035	.035		.620	−.012	−.018	−.071	−.041	−.077
.852	−.100	−.012	−.064	−.071		.750	−.069	−.053	−.077	−.111	−.112
.930	−.035	.041	−.035	.012		.850	−.017	−.024	−.047	−.064	−.077
						.950	.029	.012	−.024	[illegible]	−.065
α = 10.6°											
.032	−.006	.545	.052	.321		.010	−1.422	−.991	−.783	−.695	−.800
.053	−.185	.317	−.133	.095		.080	−1.387	−1.008	−.807	−.730	−.818
.100	−.221	.111	−.255	−.131		.130	−1.119	−1.049	−.867	−.765	−.854
.145	−.161	.029	−.220	−.119		.145	−1.107	−.997	−.837	−.765	−.842
.189	−.066	.064	−.162	−.071		.155	−1.078	−1.032	−.831	−.765	−.842
.234	−.108	.123	−.081	−.095		.180	−.991	−1.032	−.843	−.776	−.854
.280	−.078	.135	.012	−.077		.220	−.880	−1.020	−.861	−.805	−.854
.326	−.125	.129	−.023	−.065		.270	−.734	−.962	−.878	−.823	−.854
.371	−.179	.164	−.151	.018		.400	−.455	−.709	−.760	−.759	−.758
.392	−.220	.189	−.365	.237	Upper	.620	−.256	−.340	−.433	−.475	−.484
.413	−.257	.211	−.684	.332		.685					
.434	−.305	.217	−.863	.279		.693					
.457	−.299	.190	−.834	.220		.700	−.122	−.235	−.273	−.324	−.376
.480	−.299	.160	−.591	.142		.720	−.128	−.193	−.285	−.330	−.352
.502	−.328	.130	−.440	.059		.750	−.117	−.182	−.267	−.324	−.352
.551	−.215	.100	−.249	.012		.800	−.070	−.147	−.226	−.272	−.299
.585	−.179	.076	−.185	−.006		.900	−.029	−.088	−.166	−.209	−.281
.592	−.137	.106	−.151	−.012		.980	−.029	−.082	−.125	−.162	−.185
.613	−.102	.100	−.104	−.018							
.634	−.066	.100	−.058	.012		.025	.769	.721	.706	.649	.567
.655	−.054	.088	−.023	.012		.120	.472	.422	.404	.411	.376
.675	−.012	.018	.023	.024		.220	.274	.229	.208	.243	.215
.696	.012	.053	.023	.042	Lower	.300	.192	.147	.154	.162	.084
.774	−.024	.015	.035	.036		.620	.035	.006	−.024	−.012	−.108
.852	−.108	−.023	−.015	−.071		.750	−.029	−.029	−.083	−.081	−.113
.930	−.066	.076	−.070	.059		.850	−.035	−.059	−.077	−.087	−.113
						.950	.035	−.041	−.071	−.093	−.137

TABLE 23 Concluded
(e) Concluded

PRESSURE COEFFICIENTS FOR FUSELAGE, WING, FLAP, OR AILERON

Wing configuration

δ_n = 00°; δ_f = 60°; $\delta_{a,L}$ = 00°; $\delta_{a,R}$ = 00°; h_s/c = 1.0 h_d/c = 0.5

$C_{\mu,k}$ = 0.000 $C_{\mu,f}$ = 0.000 $C_{\mu,a}$ = 0.000

x/l	Fuselage, 0.000, Upper surface	Fuselage, 0.000, Lower surface	Fuselage, 0.154, Upper surface	Fuselage, 0.154, Lower surface	Surface	x/c	Wing, flap, or aileron, 0.221	Wing, flap, or aileron, 0.423	Wing, flap, or aileron, 0.640	Wing, flap, or aileron, 0.800	Wing, flap, or aileron, 0.918
α = 14.4°											
.032	-.077	.633	-.083	.251	Upper	.010	-.674	-.667	-.555	-.643	-.677
.053	-.214	.394	-.236	.048		.080	-.715	-.723	-.573	-.661	-.665
.100	-.172	.197	-.336	-.161		.130	-.663	-.741	-.597	-.672	-.677
.145	-.113	.084	-.283	-.155		.145	-.715	-.753	-.621	-.678	-.689
.189	-.059	.137	-.195	-.096		.155	-.715	-.741	-.603	-.667	-.677
.234	-.053	.191	-.094	-.119		.180	-.715	-.747	-.597	-.672	-.671
.280	-.030	.209	-.006	-.096		.220	-.704	-.753	-.627	-.672	-.671
.326	-.006	.179	-.018	-.072		.270	-.715	-.782	-.651	-.708	-.700
.371	-.095	.227	-.112	.048		.400	-.768	-.830	-.699	-.755	-.712
.392	-.145	.245	-.248	.287		.620	-.692	-.717	-.753	-.755	-.700
.413	-.190	.269	-.643	.388		.685					
.434	-.285	.269	-.672	.358		.693					
.457	-.344	.230	-.702	.287		.700	-.534	-.597	-.567	-.619	-.641
.480	-.404	.195	-.726	.191		.720	-.534	-.571	-.585	-.625	-.635
.502	-.487	.160	-.708	.119		.750	-.481	-.533	-.615	-.631	-.605
.551	-.451	.125	-.714	.006		.800	-.393	-.46[illegible]	-.603	-.613	-.594
.585	-.427	.090	-.667	-.006		.900	-.240	-.334	-.549	-.578	-.588
.592	-.398	.108	-.560	-.027		.980	-.293	-.352	-.454	-.490	-.528
.613	-.303	.090	-.437	-.048							
.634	-.267	.096	-.295	-.078	Lower	.025	.797	.776	.705	.702	.617
.655	-.220	.078	-.171	-.084		.120	.504	.496	.430	.454	.410
.675	-.113	.036	-.100	-.078		.220	.311	.293	.239	.248	.220
.696	-.059	.048	-.035	-.090		.300	.240	.191	.155	.183	.101
.774	.018	.015	.029	.012		.620	.029	[illegible]	-.072	-.047	-.137
.852	-.083	-.018	-.059	-.072		.750	-.047	-.054	-.125	-.159	-.172
.930	-.047	.119	-.065	.125		.850	-.082	-.090	-.167	-.183	-.220
						.950	-.088	-.143	-.239	-.265	-.326
α = 18.4°											
.032	-.125	.706	-.220	.173	Upper	.010	-.643	-.65[illegible]	-.549	-.576	-.611
.053	-.261	.475	-.356	-.030		.080	-.684	-.70[illegible]	-.579	-.576	-.611
.100	-.166	.261	-.433	-.227		.130	-.667	-.714	-.585	-.605	-.629
.145	-.113	.178	-.368	-.233		.145	-.761	-.73[illegible]	-.591	-.617	-.617
.189	-.024	.208	-.291	-.179		.155	-.720	-.71[illegible]	-.579	-.594	-.611
.234	.006	.237	-.113	-.173		.180	-.726	-.72[illegible]	-.591	-.594	-.623
.280	.006	.249	-.012	-.167		.220	-.720	-.73[illegible]	-.585	-.594	-.611
.326	.030	.255	.006	-.096		.270	-.737	-.76[illegible]	-.597	-.623	-.623
.371	-.077	.309	-.083	-.006		.400	-.726	-.78[illegible]	-.669	-.665	-.659
.392	-.012	.317	-.285	.287		.620	-.726	-.78[illegible]	-.836	-.772	-.706
.413	-.231	.326	-.742	.424		.685					
.434	-.344	.321	-.766	.388		.693					
.457	-.392	.270	-.801	.334		.700	-.661	-.74[illegible]	-.561	-.647	-.694
.480	-.439	.220	-.825	.245		.720	-.631	-.73[illegible]	-.615	-.635	-.694
.502	-.510	.170	-.825	.155		.750	-.584	-.70[illegible]	-.675	-.659	-.700
.551	-.528	.120	-.795	.024		.800	-.501	-.63[illegible]	-.681	-.700	-.694
.585	-.582	.089	-.712	-.030		.900	-.313	-.51[illegible]	-.621	-.694	-.671
.592	-.611	.107	-.635	-.045		.980	-.372	-.49[illegible]	-.561	-.683	-.671
.613	-.481	.089	-.499	-.060							
.634	-.505	.071	-.386	-.084	Lower	.025	.855	.80[illegible]	.776	.754	.629
.655	-.433	.047	-.279	-.096		.120	.584	.54[illegible]	.502	.505	.421
.675	-.332	-.012	-.184	-.102		.220	.378	.34[illegible]	.299	.309	.243
.696	-.243		-.119	-.113		.300	.295	.27[illegible]	.227	.237	.125
.774	-.024	-.015	-.006	-.066		.620	.053	.01[illegible]	-.060	-.053	-.172
.852	-.065	-.030	-.071	-.030		.750	-.053	-.06[illegible]	-.143	-.166	-.243
.930	-.077	.160	-.137	.155		.850	-.100	-.13[illegible]	-.215	-.267	-.309
						.950	-.136	-.24[illegible]	-.328	-.410	-.457

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TABLE 24

SAMPLE DATA SHEET FOR LANDING-FLARE CALCULATION FOR AIRPLANE WITHOUT BOUNDARY-LAYER CONTROL

[Calculations begin with equilibrium conditions along a flight path, γ]

Column number	
1	t, time, sec
2	Δt, increment of time
3	γ, flight-path angle (- for descent, + for climb, $\tan^{-1}\dot{Z}/\dot{X}$), deg
4	α, angle of attack of wing and fuselage to flight path, deg
5	W, weight of airplane, lb
6	M, mass, W/g
7	T, thrust (determined from previous 36), lb
8	$\cos\alpha$, cos 4
9	$\sin\alpha$, sin 4
10	$T\cos\alpha$
11	$T\sin\alpha$
12	C_L (trimmed force data)
13	C_D (trimmed force data)
14	L, 12 × previous 39 40
15	D, 13 × previous 39 40
16	$\cos\gamma$, cos 3
17	$\sin\gamma$, sin 3
18	$T\cos\alpha - D$, 10 - 15
19	$L + T\sin\alpha$, 14 + 11
20	$(T\cos\alpha - D)\cos\gamma$, 18 16
21	$(L + T\sin\alpha)\sin\gamma$, 19 17
22	$(T\cos\alpha - D)\cos\gamma - (L + T\sin\alpha)\sin\gamma$, 20 - 21
23	$\ddot{X}$, horizontal acceleration, 22/6, ft/sec^2
24	$(T\cos\alpha - D)\sin\gamma$, 18 17
25	$(L + T\sin\alpha)\cos\gamma$, 19 16
26	$(T\cos\alpha - D)\sin\gamma + (L + T\sin\alpha)\cos\gamma - W$, 24 + 25 - 5
27	$\ddot{Z}$, vertical acceleration, 26/6, ft/sec^2
28	$\ddot{X}\,\Delta t$, 23 2
29	$\dot{X}$, horizontal velocity, $\dot{X}_o + \ddot{X}\,\Delta t$, previous 29 + 28, ft/sec
30	$\ddot{Z}\,\Delta t$, 27 2
31	$\dot{Z}$, vertical velocity, $\dot{Z}_o + \ddot{Z}\,\Delta t$, previous 31 + 30, ft/sec
32	$\dot{X}\,\Delta t$, 29 2
33	$\ddot{X}\,\frac{\Delta t^2}{2}$, 23 × $\frac{2^2}{2}$
34	X, horizontal distance, $X_o + \dot{X}(\Delta t) + \ddot{X}\,\frac{\Delta t^2}{2}$, previous 34 + 32 + 33, ft
35	$\dot{Z}\,\Delta t$, 31 2
36	$\ddot{Z}\,\frac{\Delta t^2}{2}$, 27 × $\frac{2^2}{2}$
37	Z, vertical distance, $Z_o + \dot{Z}(\Delta t) + \ddot{Z}\,\frac{\Delta t^2}{2}$, previous 37 + 35 + 36, ft
38	V, flight-path velocity, $\sqrt{\dot{X}^2 + \dot{Z}^2}$, $\sqrt{29^2 + 31^2}$, ft/sec
39	q, flight-path dynamic pressure, $(0.001189)(V^2)$, $(0.001189)(38)^2$, lb/sq ft
40	S, wing area, sq ft

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TABLE 25

SAMPLE DATA SHEET FOR LANDING-FLARE CALCULATION FOR AIRPLANE

WITH BOUNDARY-LAYER CONTROL

[All data involving C_μ apply to this particular airplane only; calculations begin with equilibrium conditions along a flight path, γ]

Column number		Column number	
(1)	t, time, sec	(23)	$L + T \sin \alpha$, (16) + (9)
(2)	Δt, increment of time	(24)	$(T \cos \alpha - D)\cos \gamma$, (22) (18)
(3)	γ, flight-path angle (- for descent, + for climb, $\tan^{-1}\dot{Z}/\dot{X}$), deg	(25)	$(L + T \sin \alpha)\sin \gamma$, (23) (19)
(4)	α, angle of attack of wing to flight path, deg	(26)	$(T \cos \alpha - D)\cos \gamma - (L + T \sin \alpha)\sin \gamma$, (24) - (25)
(5)	W, weight of airplane, lb	(27)	$\ddot{X}$, horizontal acceleration, (26)/(6), ft/sec^2
(6)	M, mass, W/g	(28)	$(T \cos \alpha - D)\sin \gamma$, (22) (19)
(7)	T, thrust (determined from previous (42)), lb	(29)	$(L + T \sin \alpha)\cos \gamma$, (23) (18)
(8)	$T \cos \alpha$	(30)	$(T \cos \alpha - D)\sin \gamma + (L + T \sin \alpha)\cos \gamma - W$, (28) + (29) - (5)
(9)	$T \sin \alpha$	(31)	$\ddot{Z}$, vertical acceleration, (30)/(6), ft/sec^2
(10)	C_μ = 0.825/previous (43) (based on required air flow at approach speed)	(32)	$\ddot{X}\,\Delta t$, (27) (2)
(11)	ΔC_μ = (10) - previous (10)	(33)	$\dot{X}$, horizontal velocity, $\dot{X}_o + \ddot{X}\,\Delta t$, previous (3) + (32), ft/sec
(12)	ΔC_L = 13.1 (11)	(34)	$\ddot{Z}\,\Delta t$, (31) (2)
(13)	C_L (trimmed force data)	(35)	$\dot{Z}$, vertical velocity, $\dot{Z}_o + \ddot{Z}\,\Delta t$, previous (35) + (34), ft/sec
(14)	$C_{L,tot}$ = (13) + (12)	(36)	$\dot{X}\,\Delta t$, (33) (2)
(15)	C_D (trimmed force data)	(37)	$\ddot{X}\,\frac{\Delta t^2}{2}$, (27) $\frac{(2)^2}{2}$
(16)	L, (14) x previous (43) (44)	(38)	X, horizontal distance, $X_o + \dot{X}\,\Delta t + \ddot{X}\,\frac{\Delta t^2}{2}$, previous (38) + (36) + (37), ft
(17)	D, (15) x previous (43) (44)	(39)	$\dot{Z}\,\Delta t$, (35) (2)
(18)	$\cos \gamma$, cos (3)	(40)	$\ddot{Z}\,\frac{\Delta t^2}{2}$, (31) $\frac{(2)^2}{2}$
(19)	$\sin \gamma$, sin (3)	(41)	Z, vertical distance, $Z_o + \dot{Z}\,\Delta t + \ddot{Z}\,\frac{\Delta t^2}{2}$, previous (41) + (39) + (40), ft
(20)	$\cos \alpha$, cos (4)	(42)	V, flight-path velocity, $\sqrt{\dot{X}^2 + \dot{Z}^2}$, $\sqrt{(33)^2 + (35)^2}$, ft/sec
(21)	$\sin \alpha$, sin (4)	(43)	q, flight-path dynamic pressure, $(0.001189)(V^2)$, $(0.001189)((42))^2$, lb/sq ft
(22)	$T \cos \alpha - D$, (8) - (17)	(44)	S, wing area, sq ft

TABLE 26

LANDING-FLARE CALCULATIONS

[$\delta_f = 37^\circ$; $\delta_a = 37^\circ$; $\delta_n = 30^\circ$; no boundary-layer control]

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22
t, sec	Δt	γ, $\tan^{-1}$ prev. $\dot{Z}/\dot{X}$	α, deg	W, lb	M, W/g	T, lb (from prev. vel.)	Cos α	Sin α	T cos α	T sin α	C_L (trim data)	C_D (trim data)	L, 12 40 × prev. 39	D, 13 40 × prev. 39	Cos γ	Sin γ	10 - 15	14 + 11	18 16	19 17	20 - 21
0	0	-3.00	5.30	18,000	558	5,152	0.996	0.092	5,131	474	0.813	0.282	17,526	6,079	0.999	-0.052	-948	18,000	-947	-936	-11
1	1	-3.00	5.85			5,152	.995	.102	5,126	526	.838	.288	18,101	6,221		-.052	-1,095	18,627	-1,094	-969	-125
2	1	-2.72	6.40			5,152	.994	.111	5,121	572	.868	.295	18,723	6,363		-.047	-1,242	19,295	-1,241	-907	-334
3	1	-2.17	6.95			5,152	.993	.121	5,116	623	.898	.303	19,262	6,499		-.038	-1,383	19,885	-1,382	-756	-626
4	1	-1.37	6.95			0	.993	.121	0	0	.898	.303	19,074	6,436	1.000	-.024	-6,436	19,074	-6,436	-458	-5,978
5	1	-.90	7.95			0	.990	.138	0	0	.947	.312	18,381	6,056		-.016	-6,056	18,381	-6,056	-294	-5,762
6	1	-.72	9.70			0	.986	.168	0	0	1.038	.338	18,404	5,993		-.013	-5,993	18,404	-5,993	-239	-5,754

23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
$\ddot{X}$, 22/6	18 17	19 16	24 + 25 - 5	$\ddot{Z}$, 26/6	23 2	$\dot{X}$, prev. 29 + 28	27 2	$\dot{Z}$, prev. 31 + 30	29 2	23 $\frac{2^2}{2}$	X, prev. 34 + 32 + 33	31 2	27 $\frac{2^2}{2}$	Z, prev. 37 + 35 + 36	V, $\sqrt{29^2 + 31^2}$, ft/sec	q, 0.001189 38^2	S
-0.02	49.3	17,982	31.3	0.06	0	245.8	0	-12.87	0	0	0	0	0	0	246.1	72.0	300
-.22	56.9	18,608	664.9	1.19	-.22	245.6	1.19	-11.68	245.6	-.11	245.5	-11.68	.60	-11.08	245.9	71.9	
-.60	58.4	19,276	1,334.4	2.39	-.60	245.0	2.39	-9.29	245.0	-.30	490.2	-9.29	1.20	-19.17	245.2	71.5	
-1.12	52.6	19,865	1,917.6	3.44	-1.12	243.9	3.44	-5.85	243.9	-.56	733.5	-5.85	1.72	-23.30	244.0	70.8	
-10.71	154.5	19,074	1,228.5	2.20	-10.71	233.2	2.20	-3.65	233.2	-5.36	961.3	-3.65	1.10	-25.85	233.2	64.7	
-10.33	96.9	18,381	477.9	.86	-10.33	222.9	.86	-2.79	222.9	-5.17	1,179.0	-2.79	.43	-28.21	222.9	59.1	
-10.31	77.9	18,404	481.9	.86	-10.31	212.6	.86	-1.93	212.6	-5.16	1,386.4	-1.93	.43	-29.71	212.6	53.7	

TABLE 27

LANDING-FLARE CALCULATIONS

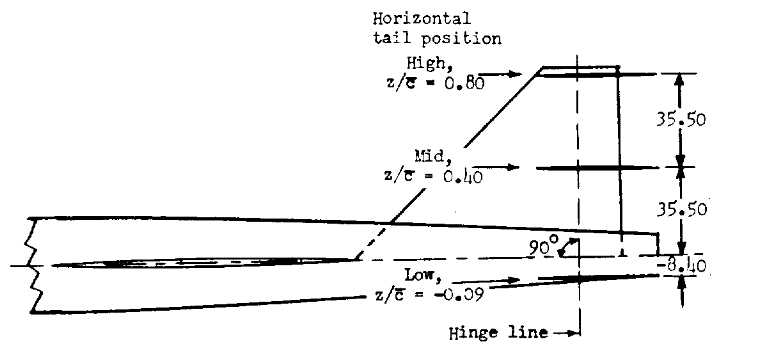
$[\delta_f = 47^\circ;\ \delta_a = 47^\circ;\ \delta_n = 50^\circ;\ C_{\mu,f} = 0.012;\ C_{\mu,a} = 0.004;\ C_{\mu,k} = 0.010]$

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)	(16)	(17)	(18)	(19)	(20)	(21)	(22)	(23)	(24)	(25)
t, sec	Δt	γ, $\tan^{-1}$ $\dot{Z}/\dot{X}$ prev.	α, deg	W, lb	M, W/g	T, lb (from prev. vel.)	T cos α	T sin α	C_μ, $\frac{0.825}{\text{prev.}\ (43)}$	ΔC_μ, (10) - prev. (10)	ΔC_L, $13.1\Delta C_\mu$	C_L (trim data)	$C_{L,tot}$, (13) + (12)	C_D (trim data)	L, (14) (44) × prev. (43)	D, (15) (44) × prev. (43)	Cos γ	Sin γ	Cos α	Sin α	(8) - (17)	(16) + (9)	(22) (18)	(23) (19)
0	0	-3.00	0.60	18,000	558	4,850	4,850	48.5	0.0162	0	0	1.178	1.178	0.380	17,917	5,780	0.999	-0.052	1.000	0.010	-930	17,966	-929	-934
1	1	-3.00	.85	↓	↓	4,850	4,850	72.8	.0163	.0001	.002	1.195	1.197	.430	18,206	6,540	↓	-.052	↓	.015	-1,690	18,279	-1,688	-951
2	1	-2.85	1.10			4,860	4,860	92.3	.0164	.0001	.002	1.215	1.217	.440	18,328	6,626		-.050		.019	-1,766	18,420	-1,764	-921
3	1	-2.62	1.35			4,860	4,860	116.6	.0167	.0003	.004	1.237	1.241	.455	18,406	6,669		-.046		.024	-1,809	18,523	-1,807	-852
4	1	-2.35	1.60			4,870	4,870	136.4	.0170	.0003	.004	1.255	1.259	.466	18,371	6,794		-.041		.028	-1,924	18,507	-1,922	-759
5	1	-2.08	2.00			4,870	4,865	170.5	.0173	.0003	.004	1.285	1.289	.478	18,421	6,826		-.036	.999	.035	-1,961	18,592	-1,959	-669
6	1	-1.77	2.50			4,880	4,875	214.7	.0177	.0004	.006	1.320	1.326	.490	18,512	6,856	1.000	-.031	↓	.044	-1,961	18,727	-1,961	-581
7	1	-1.38	3.25			4,880	4,870	278.2	.0182	.0005	.006	1.370	1.376	.510	18,727	6,931	↓	-.024	.998	.057	-2,061	19,005	-2,061	-456
8	1	-.84	3.50			4,890	4,880	298.3	.0188	.0006	.008	1.385	1.393	.516	18,372	6,796		-.015	↓	.061	-1,916	18,670	-1,916	-280
9	1	-.48	3.55			4,900	4,890	303.8	.0194	.0006	.008	1.386	1.394	.517	17,841	6,607		-.008		.062	-1,717	18,145	-1,717	-145
10	1	-.40	4.25			4,910	4,895	363.3	.0199	.0005	.006	1.425	1.431	.531	17,798	6,595		-.007	.997	.074	-1,700	18,161	-1,700	-127
11	1	-.31	5.00	↓	↓	4,920	4,900	428.0	.0206	.0007	.010	1.470	1.480	.551	17,828	6,629	↓	-.005	.996	.087	-1,729	18,256	-1,729	-91

(26)	(27)	(28)	(29)	(30)	(31)	(32)	(33)	(34)	(35)	(36)	(37)	(38)	(39)	(40)	(41)	(42)	(43)	(44)
(24) - (25)	$\ddot{X}$, $\frac{(26)}{559}$	(22) (19)	(23) (18)	(28) + (29) - (5)	$\ddot{Z}$, $\frac{(30)}{559}$	(27) (2)	X, prev. (33) + (32)	(31) (2)	Z, prev. (35) + (34)	(33) (2)	(27) $\frac{(2)^2}{2}$	X, prev. (38) + (36) + (37)	(35) (2)	(31) $\frac{(2)^2}{2}$	Z, prev. (41) + (39) + (40)	V, $\sqrt{(33)^2 + (35)^2}$, ft/sec	q, 0.001189 $(42)^2$	S
5	-0.01	48.4	17,948	-3.6	-0.01	0	206.5	0	-10.83	0	0	0	0	0	0	207.0	50.7	300
-737	-1.32	87.9	18,261	348.9	.62	-1.32	205.2	.62	-10.21	205.2	-.66	204.5	-10.21	.31	-.90	205.5	50.2	↓
-843	-1.51	88.3	18,402	490.3	.88	-1.51	203.7	.88	-9.33	203.7	-.76	407.4	-9.33	.44	-18.79	203.9	49.4	
-955	-1.71	83.2	18,504	587.2	1.05	-1.71	202.0	1.05	-8.28	202.0	-.86	608.5	-8.28	.53	-26.54	202.2	48.6	
-1,163	-2.08	78.9	18,488	566.9	1.01	-2.08	199.9	1.01	-7.27	199.9	-1.04	807.4	-7.27	.51	-33.30	200.0	47.6	
-1,290	-2.31	70.6	18,573	643.6	1.15	-2.31	197.6	1.15	-6.12	197.6	-1.16	1,003.8	-6.12	.58	-38.84	197.7	46.5	
-1,380	-2.47	60.8	18,727	787.8	1.41	-2.47	195.1	1.41	-4.71	195.1	-1.24	1,197.7	-4.71	.71	-42.84	195.2	45.3	
-1,605	-2.87	49.5	19,005	1,054.5	1.89	-2.87	192.2	1.89	-2.82	192.2	-1.44	1,388.5	-2.82	.95	-44.71	192.2	43.9	
-1,636	-2.93	28.7	18,670	698.7	1.25	-2.93	189.3	1.25	-1.57	189.3	-1.47	1,576.3	-1.57	.63	-45.65	189.3	42.6	
-1,572	-2.81	13.7	18,145	158.7	.28	-2.81	186.5	.28	-1.29	186.5	-1.41	1,761.4	-1.29	.14	-46.80	186.5	41.4	
-1,573	-2.81	11.9	18,161	172.9	.31	-2.81	183.7	.31	-.98	183.7	-1.41	1,943.7	-.98	.16	-47.62	183.7	40.1	
-1,638	-2.93	8.6	18,256	264.6	.47	-2.93	180.8	.47	-.51	180.8	-1.47	2,123.0	-.51	.24	-47.89	180.8	38.9	↓

Geometric Characteristics

Airfoil section:	
Wing	0.04c, Biconvex
Horizontal tail	0.04c, Biconvex
Aspect ratio:	
Wing	2.86
Horizontal tail	3.33
Taper ratio:	
Wing	0.40
Horizontal tail	0.50
Area:	
Wing	140.00 ft^2
Horizontal tail	30.00 ft^2
Flap chord length:	
Leading edge	0.15c
Flap or aileron	0.30c
Horizontal tail length:	1.87$\bar{c}$
Spoiler-deflector chord length:	
Spoiler	0.08c
Deflector	0.07c



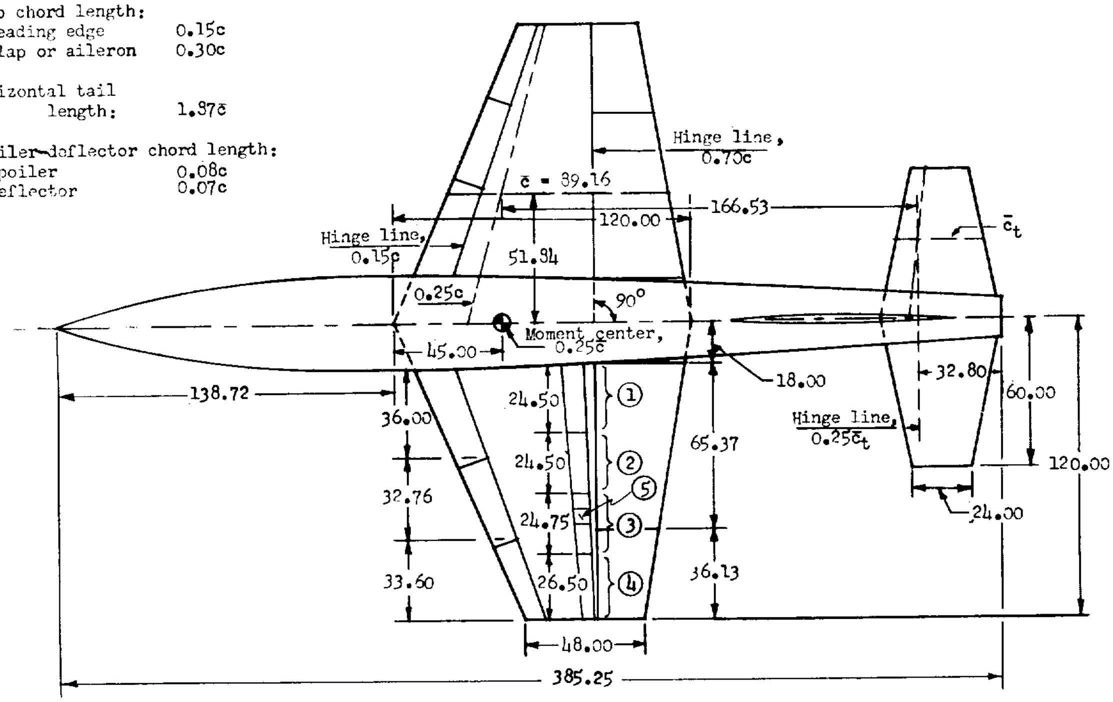
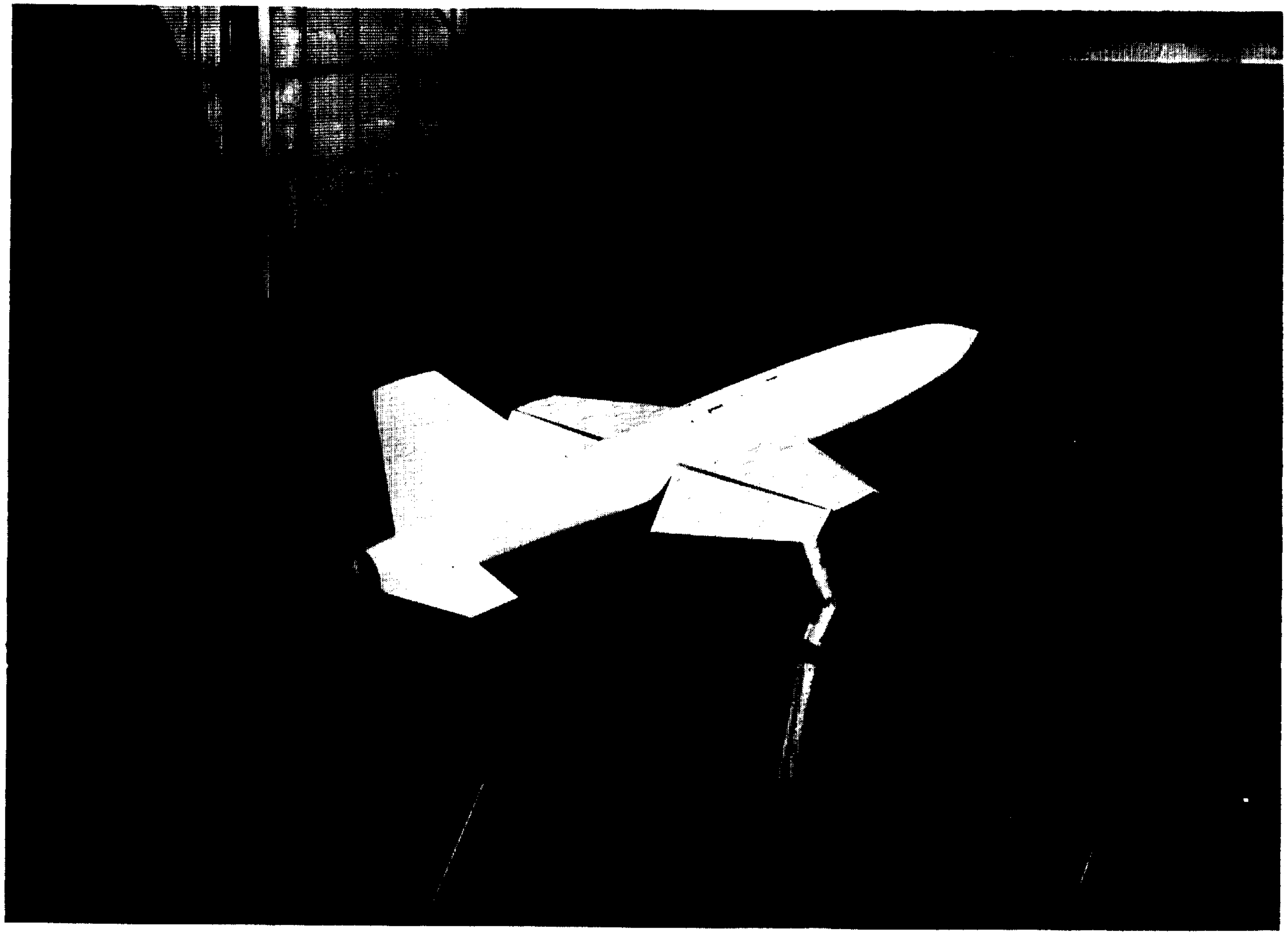
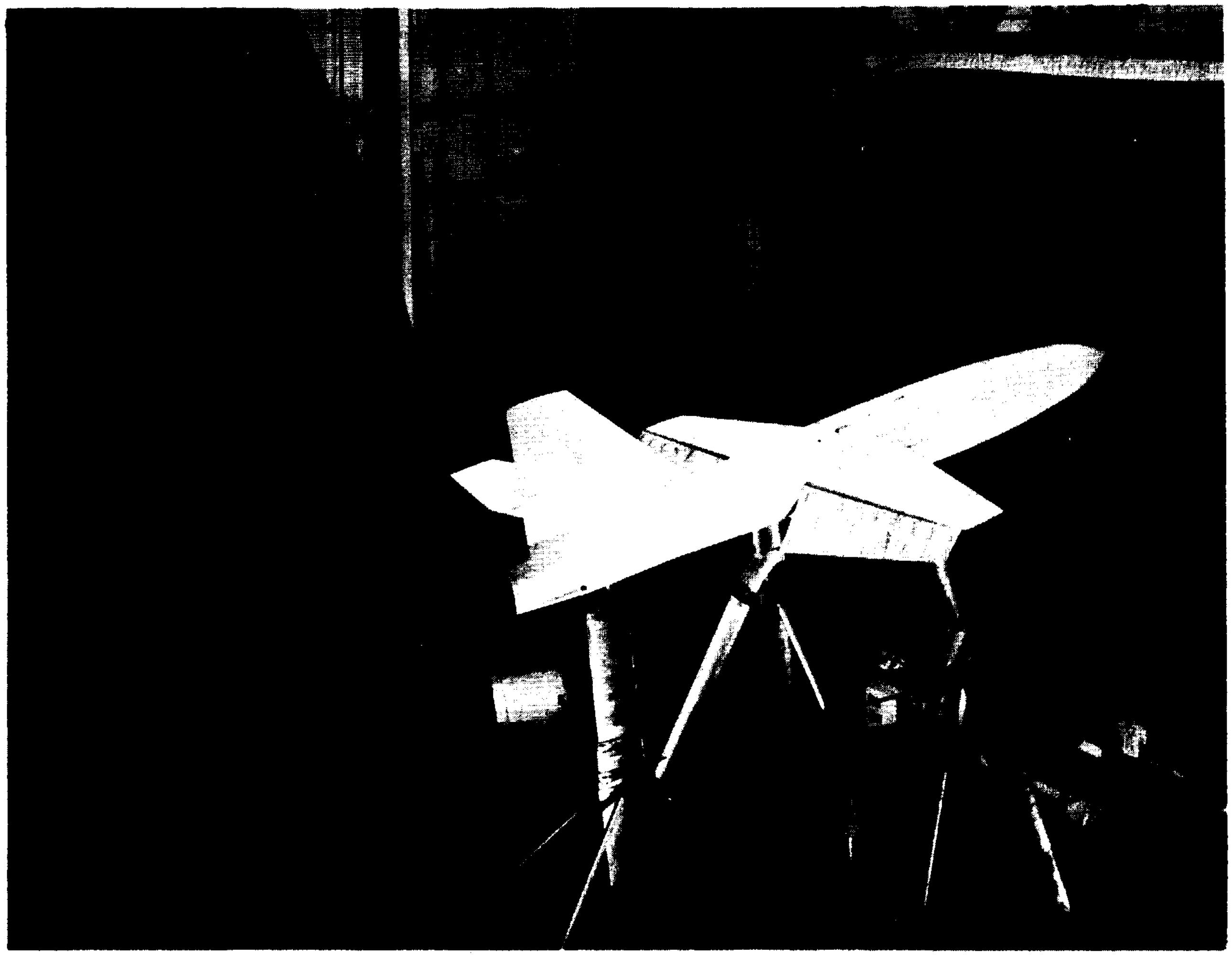


Figure 1.- Geometric characteristics of the model. All dimensions are in inches.



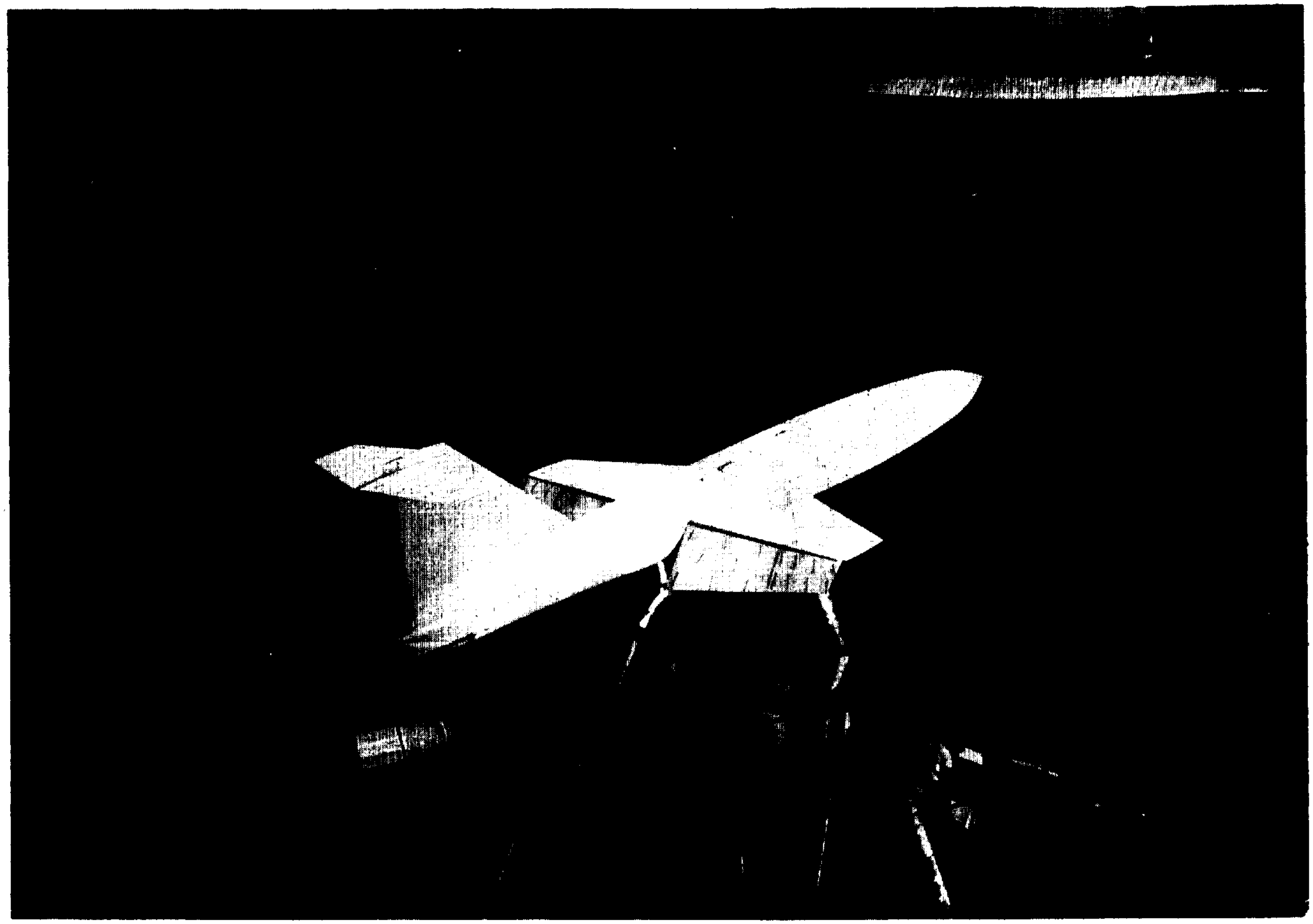
(a) 3/4 rear view of low-tail configuration. L-58-3983

Figure 2.- Photographs of the model mounted for tests in the Langley full-scale tunnel.



(b) 3/4 rear view of midtail configuration. L-58-4210

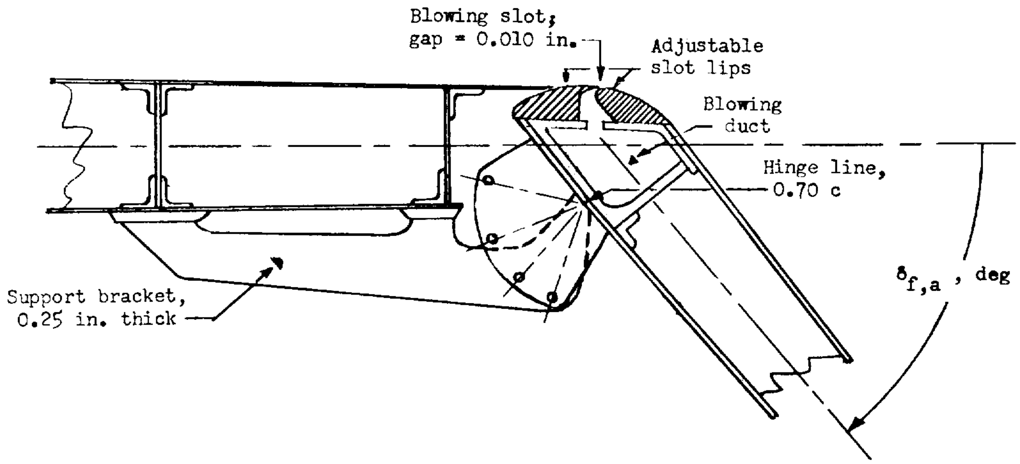
Figure 2.- Continued.



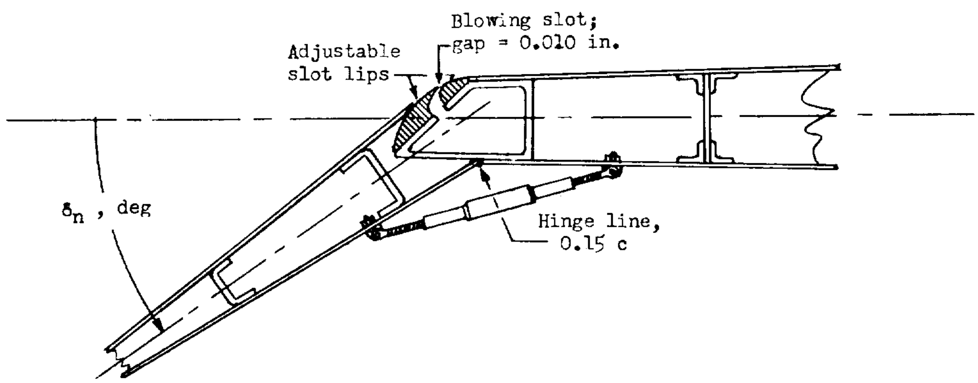
(c) 3/4 rear view of high-tail configuration. L-58-4286

Figure 2.- Concluded.

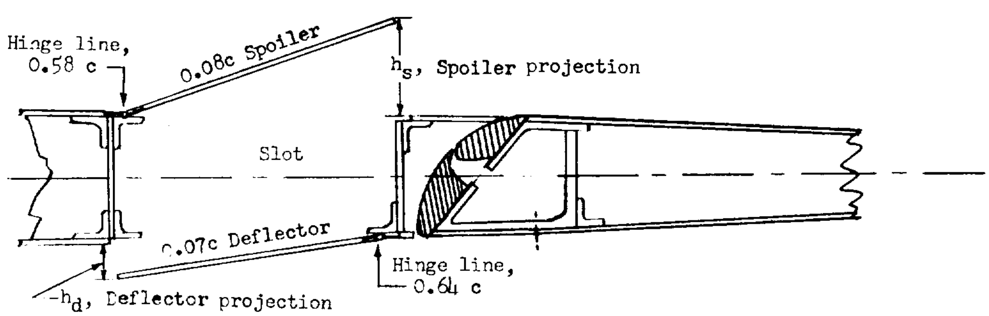
L-927



(a) Flap or aileron.



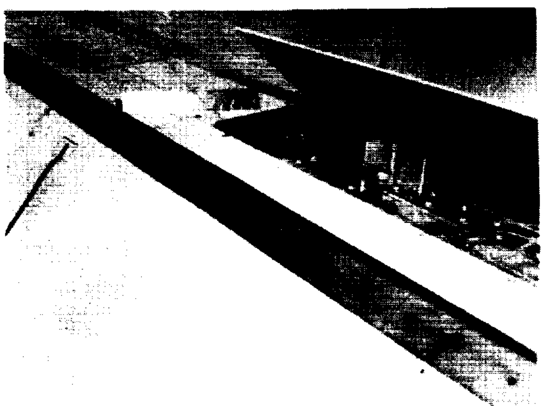
(b) Leading-edge flap.



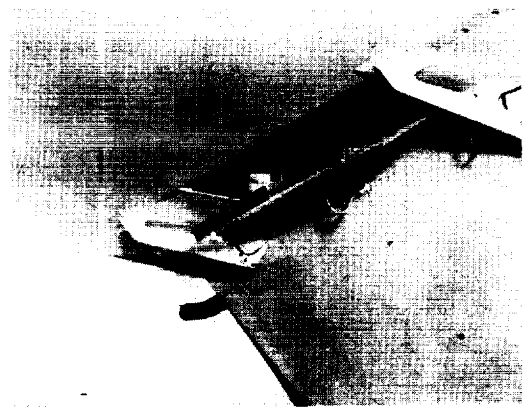
(c) Spoiler-slot-deflector.

Figure 3.- Details of flap or aileron, wing leading-edge flap, and spoiler-slot-deflector.

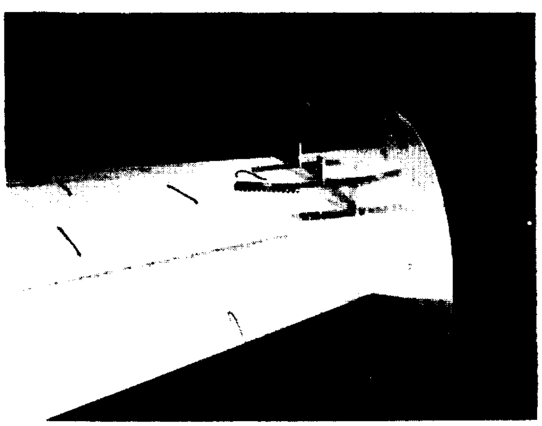
L-927



(a) Spoiler.



(b) Deflector.



(c) 3/4 front view of end plate.

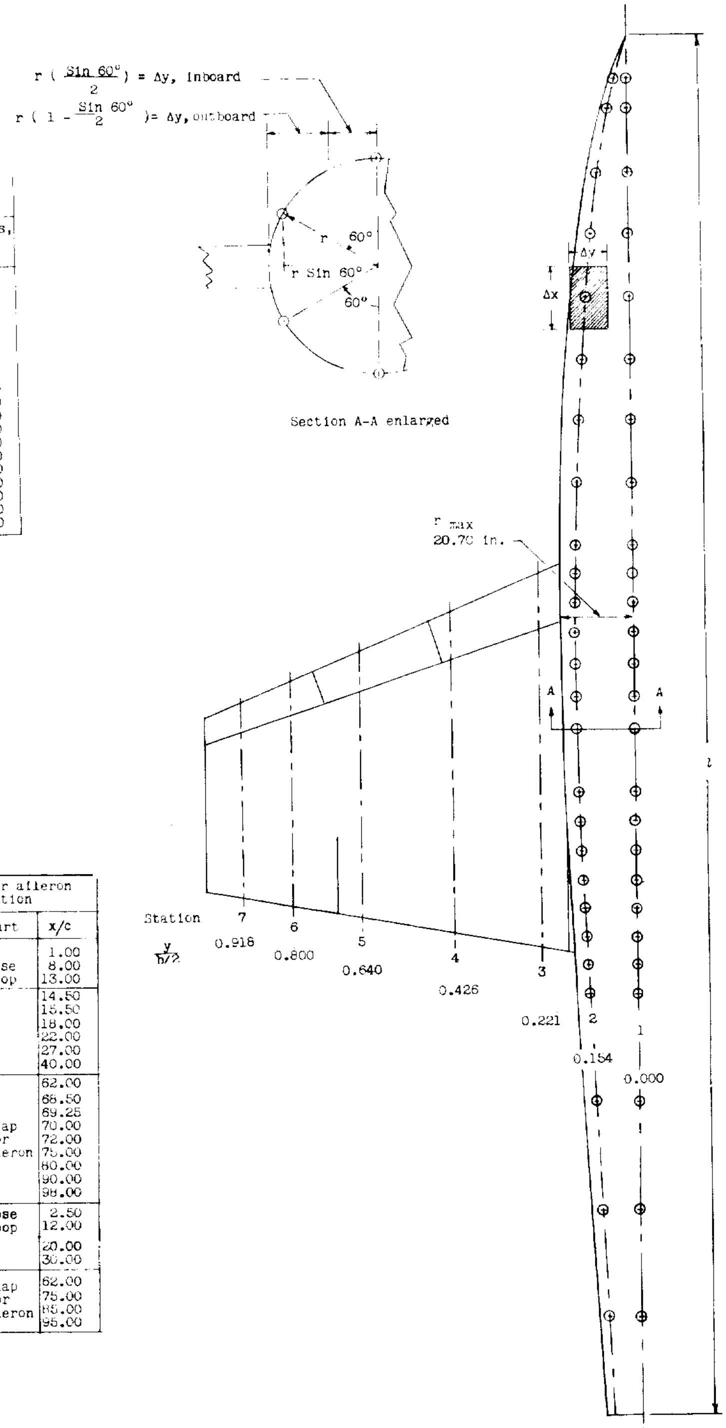
(d) 3/4 rear view of end plate.

L-60-293

Figure 4.- Photographs of spoiler, deflector, and end plate.

L-927

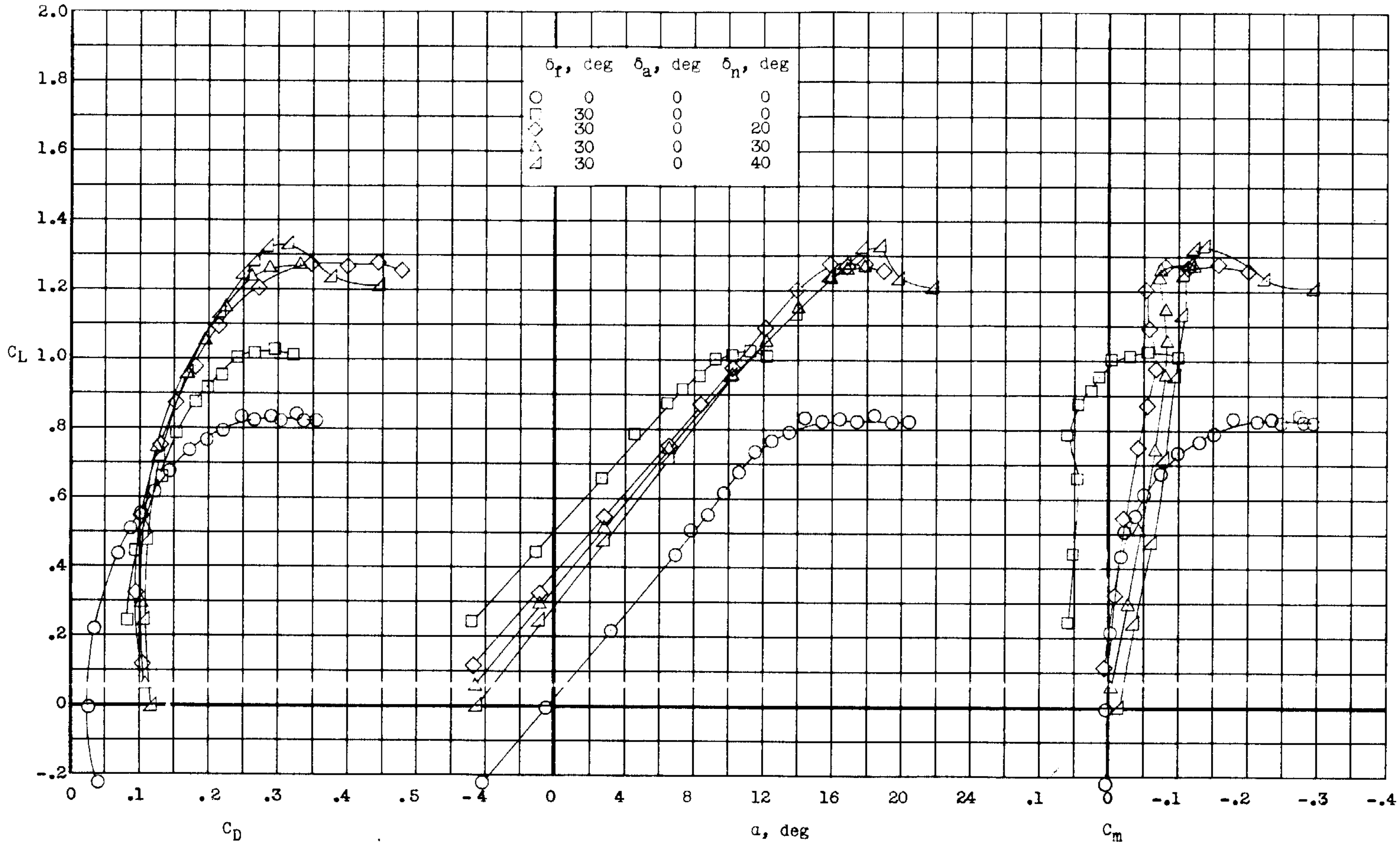
Fuselage ordinates	
Longitudinal distance, in.	Radius, in.
0.00	0.00
10.00	4.00
20.00	7.20
40.00	12.00
60.00	15.00
80.00	17.20
100.00	18.80
120.00	19.50
140.00	20.50
160.00	20.70
180.00	20.50
200.00	19.90
220.00	19.20
260.00	17.20
300.00	15.00
340.00	13.00
360.00	12.00
380.00	11.00
385.25	10.80



Fuselage orifice location, x/l
0.032
.053
.100
.145
.189
.234
.280
.326
.371
.392
.413
.434
.457
.480
.502
.551
.585
.592
.613
.634
.655
.675
.696
.774
.852
.930
l = 385.25 in.

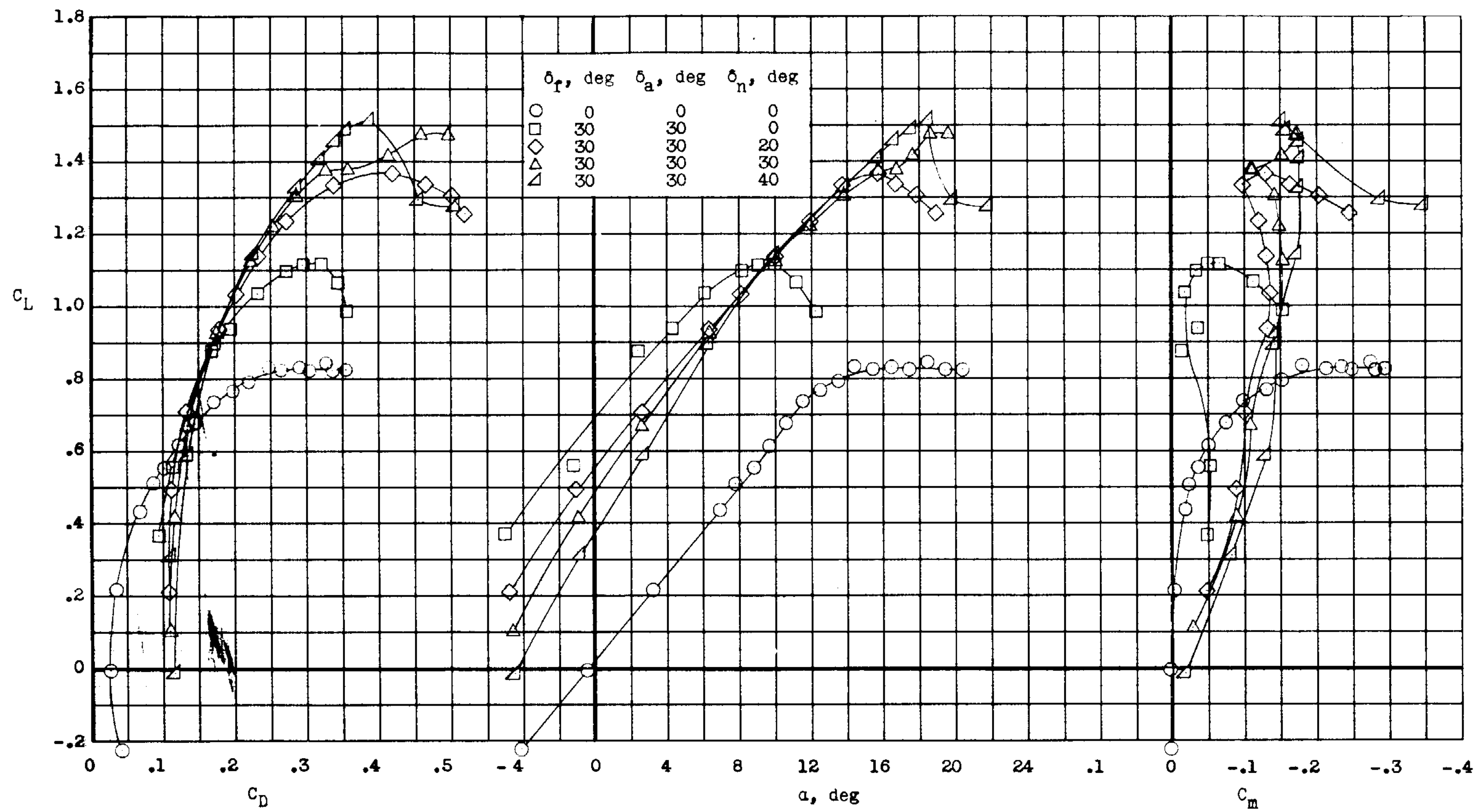
Wing, flap or aileron orifice location		
Surface	Part	x/c
Upper	Nose droop	1.00
		8.00
		13.00
		14.50
		15.50
		18.00
		22.00
		27.00
		40.00
	Flap or aileron	62.00
		68.50
		69.25
		70.00
		72.00
		75.00
		80.00
		90.00
		98.00
Lower	Nose droop	2.50
		12.00
		20.00
		30.00
	Flap or aileron	62.00
		75.00
		85.00
		95.00

Figure 5.- Fuselage and wing-surface pressure-orifice locations and fuselage ordinates.



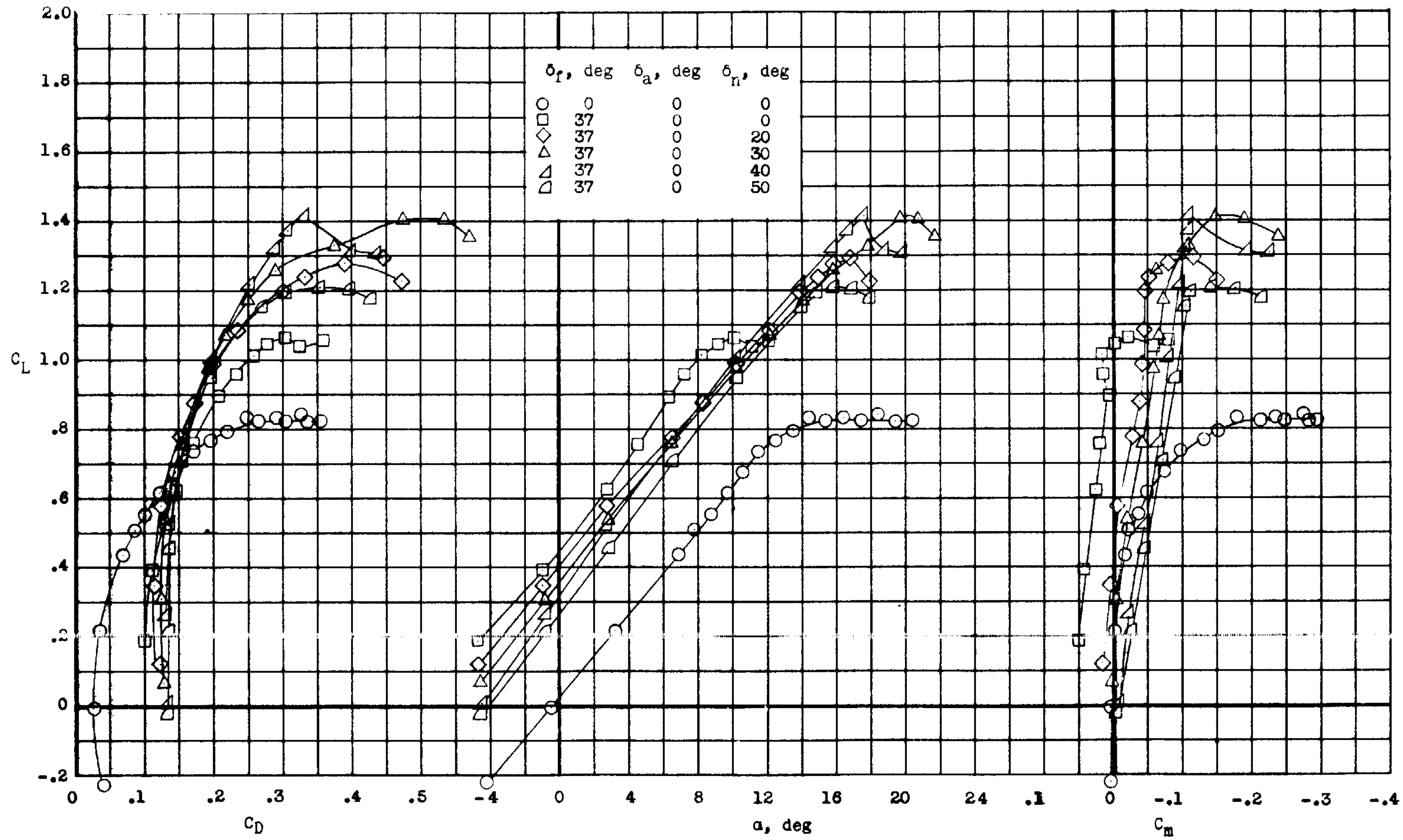
(a) Half-span flaps deflected 30°.

Figure 6.- Comparison of the effects of wing leading-edge droop on the longitudinal characteristics with either half- or full-span flaps deflected 30° to 47°, $i_t = 0°$. No boundary-layer control. $z/\bar{c} = -0.09$.



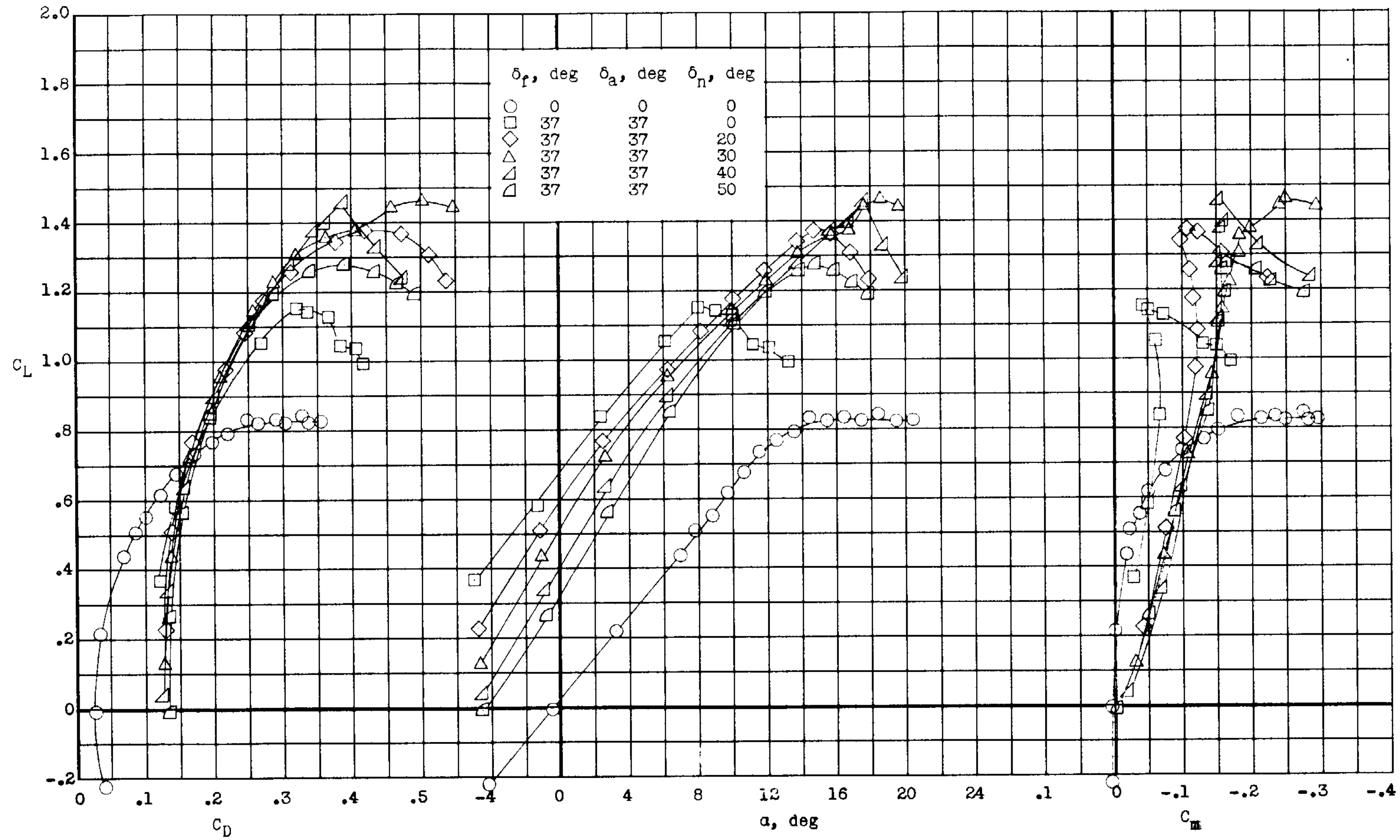
(b) Full-span flaps deflected 30°.

Figure 6.- Continued.



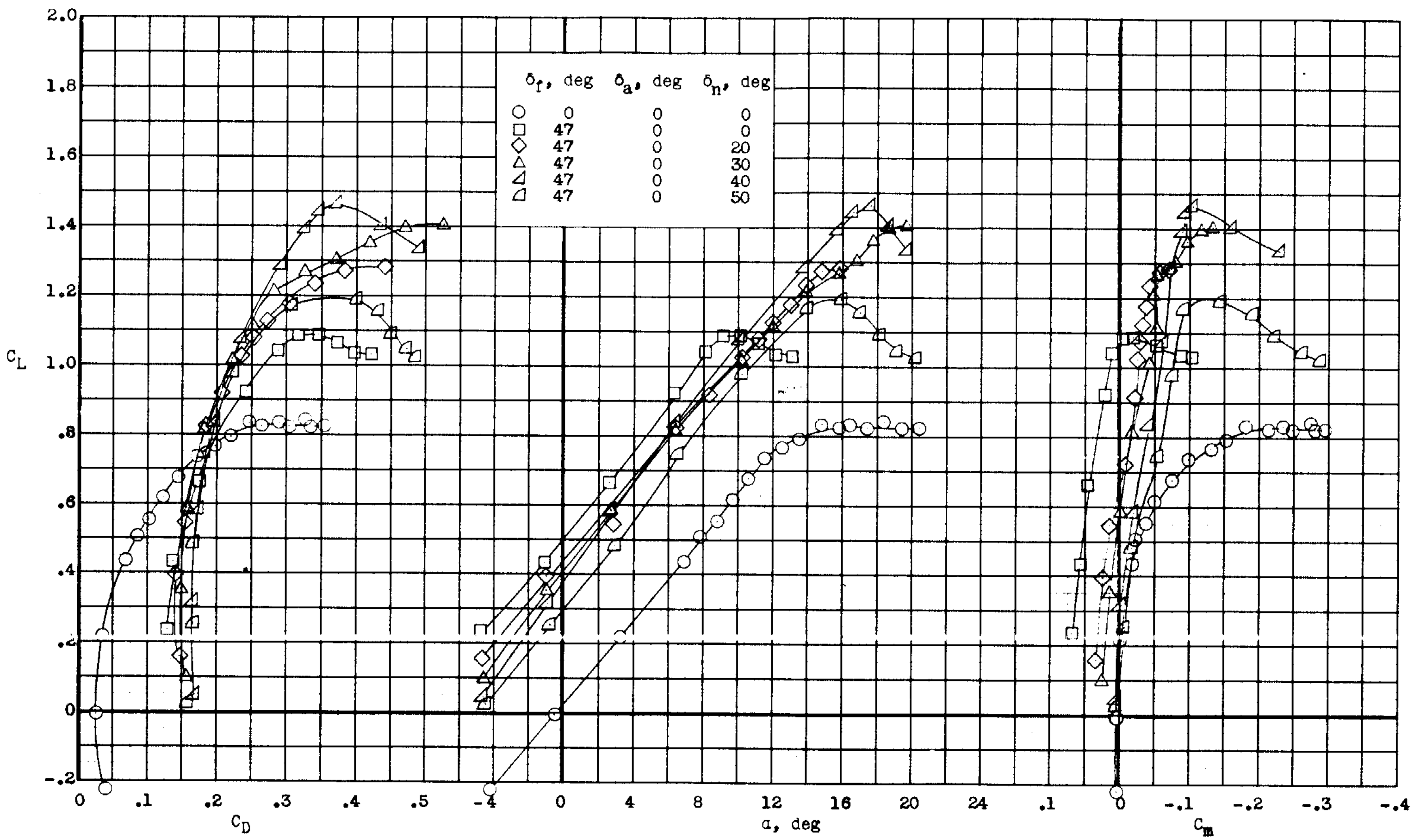
(c) Half-span flaps deflected 37°.

Figure 6.- Continued.



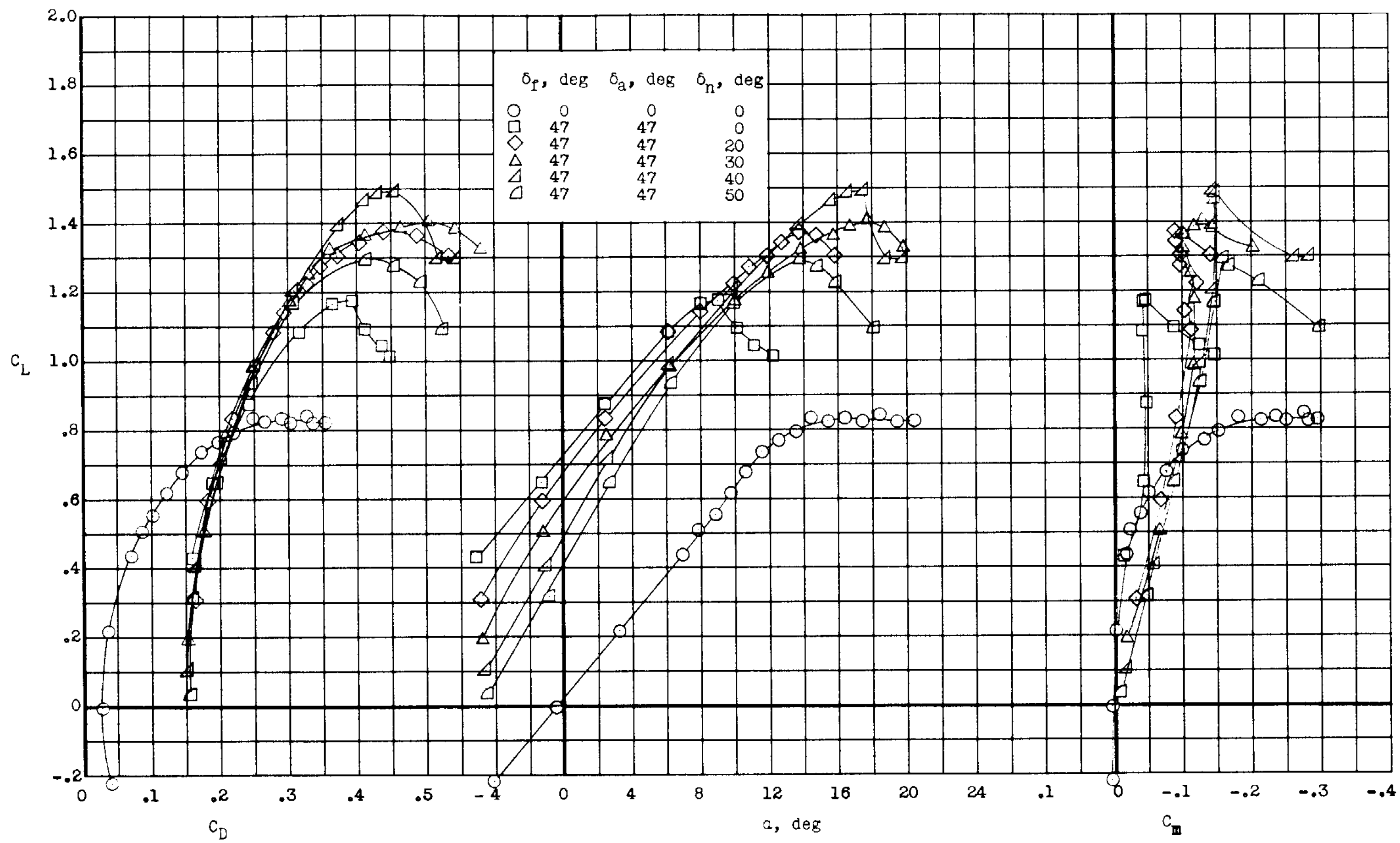
(d) Full-span flaps deflected 37°.

Figure 6.- Continued.



(e) Half-span flaps deflected 47°.

Figure 6.- Continued.



(f) Full-span flaps deflected 47°.

Figure 6.- Concluded.

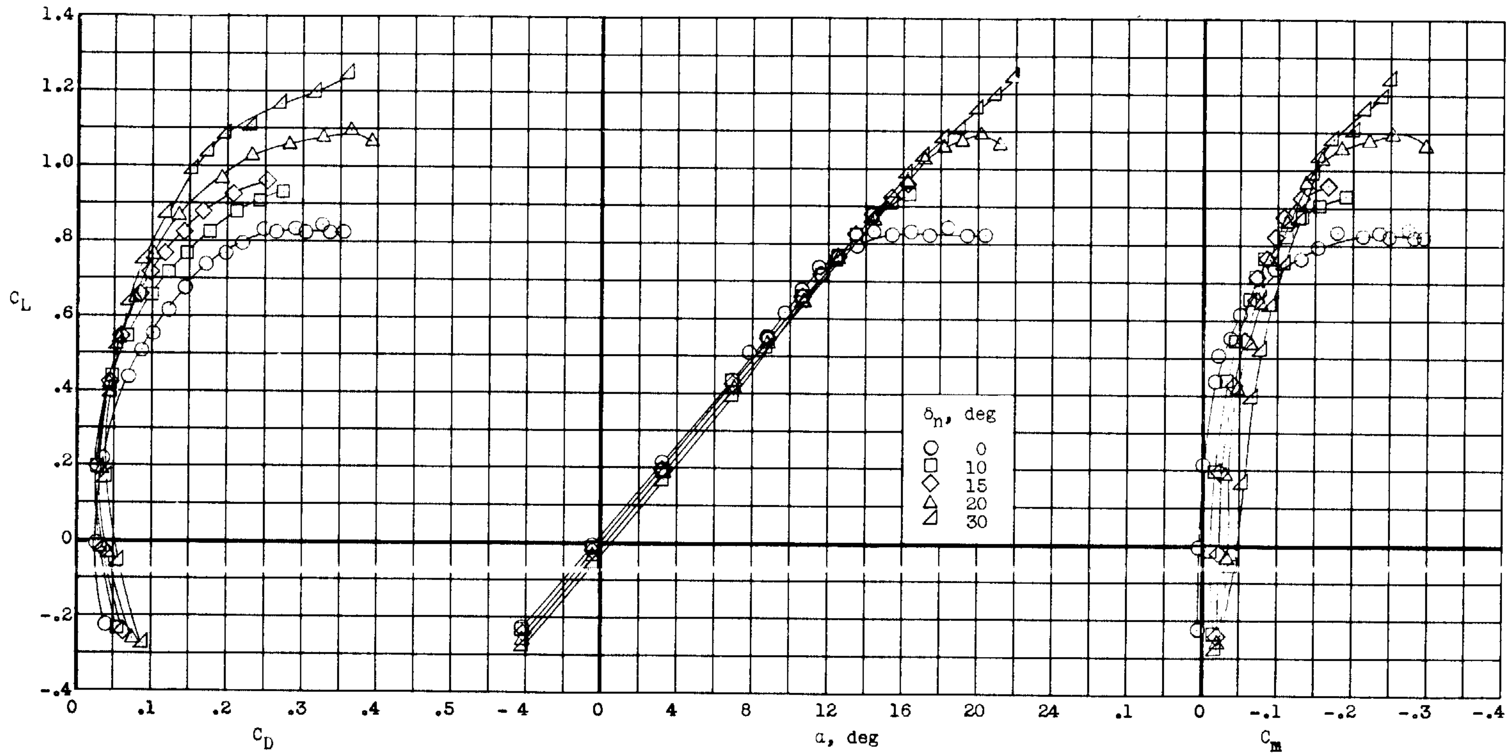
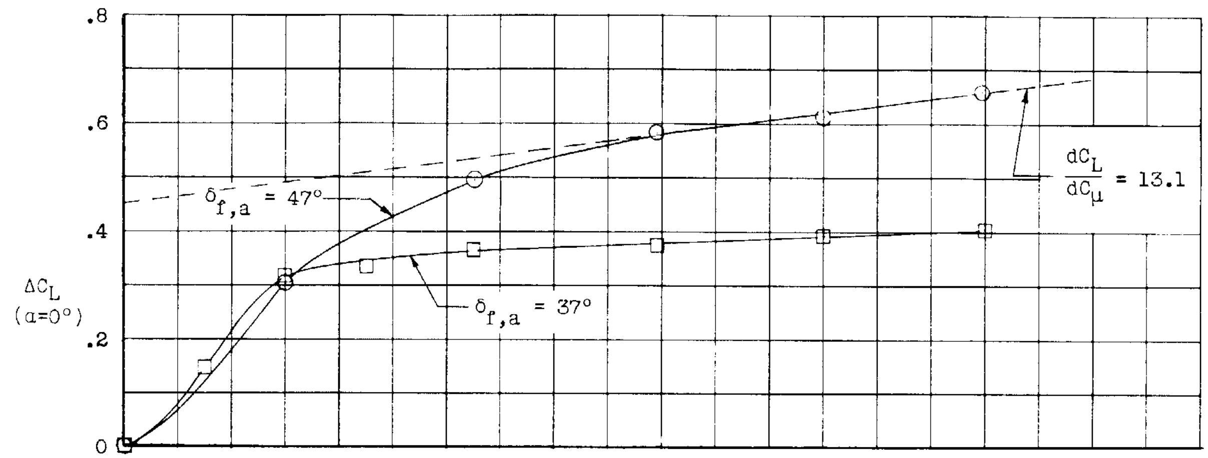
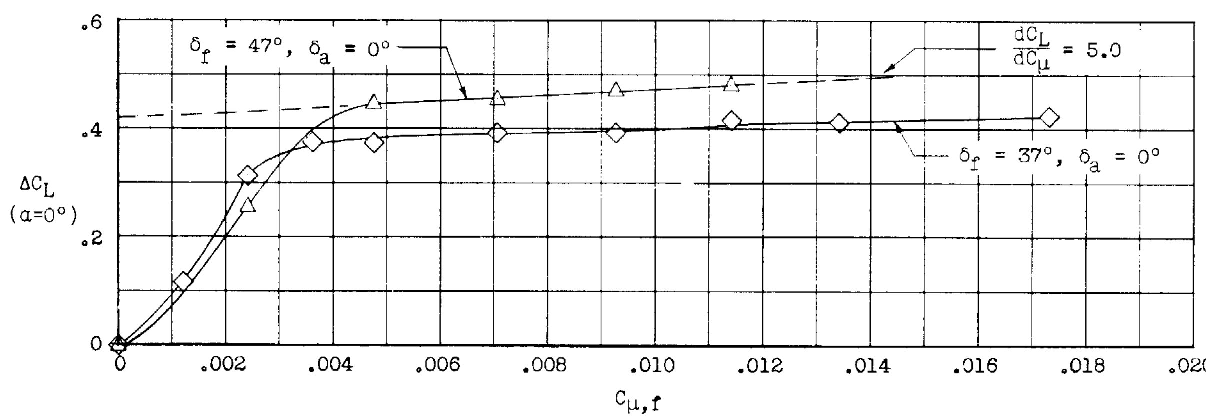


Figure 7.- Variation of the longitudinal characteristics with wing leading-edge flap deflection. No boundary-layer control. Flaps and ailerons neutral. $i_t = 0^{\circ}$. $z/\bar{c} = -0.09$.



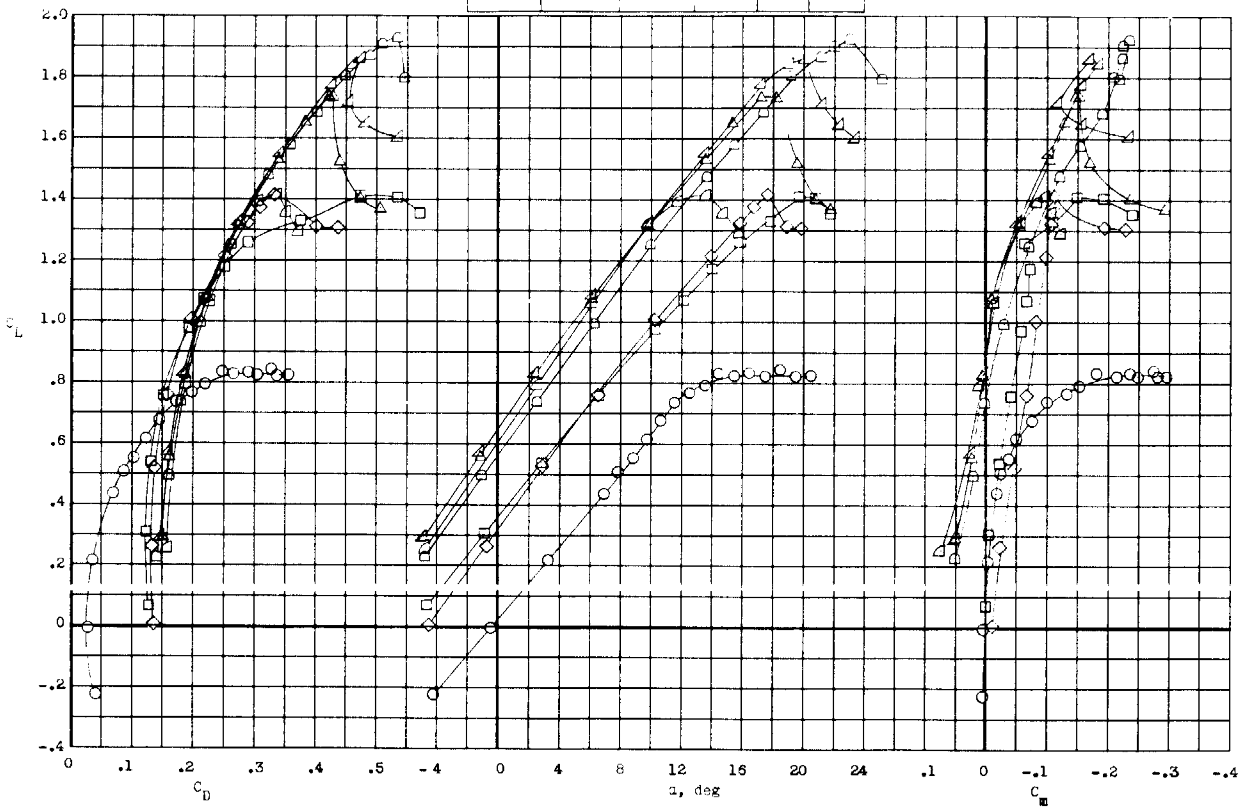
(a) Full-span flap.



(b) Half-span flap.

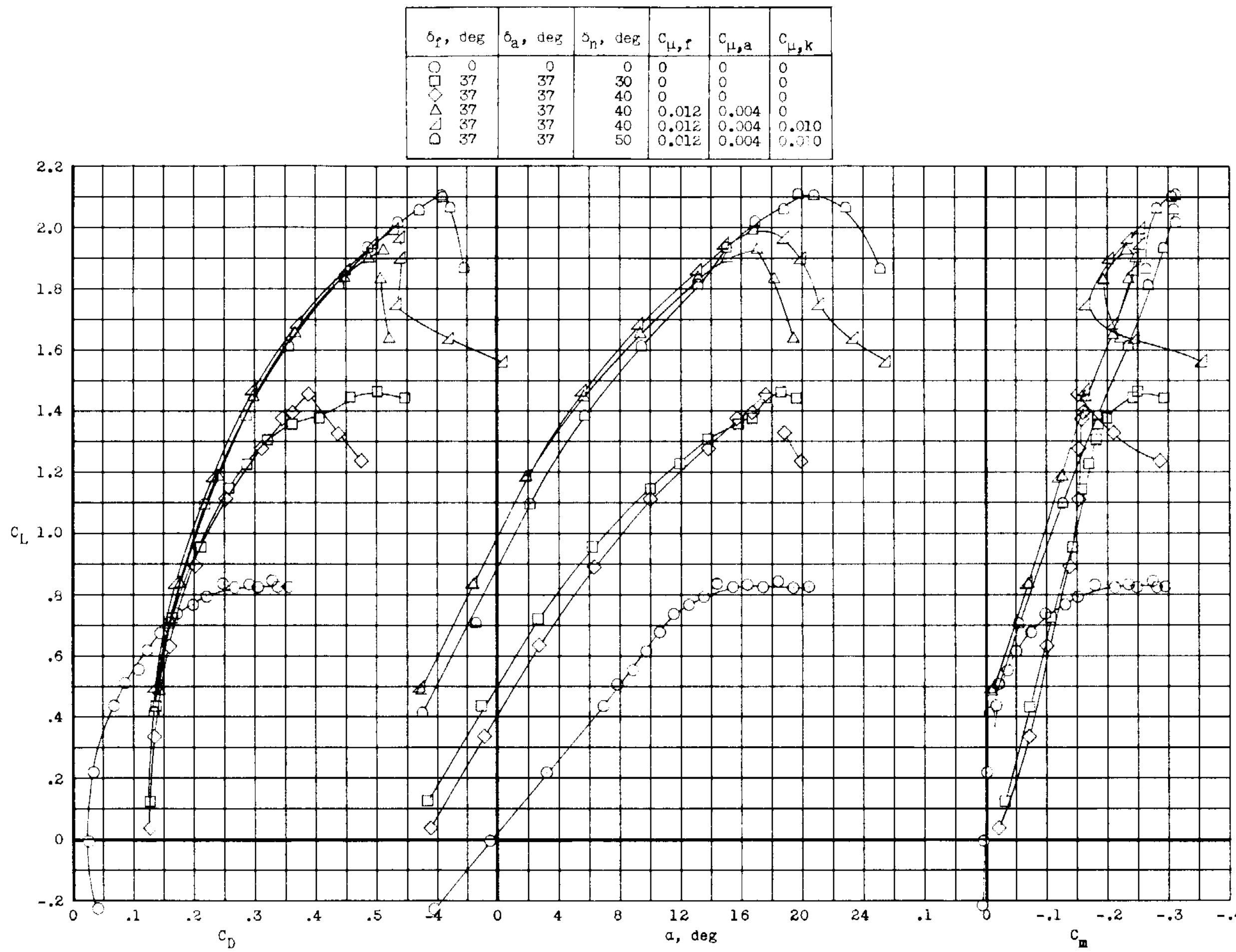
Figure 8.- Variation of incremental values of lift coefficient with flap blowing momentum coefficient for half- and full-span flap coefficients of 37° and 47°.

δ_f, deg	δ_a, deg	δ_n, deg	$C_{\mu,f}$	$C_{\mu,a}$	$C_{\mu,k}$
○ 0	0	0	0	0	0
□ 37	0	30	0	0	0
◇ 37	0	40	0	0	0
△ 37	0	40	0.012	0	0
◺ 37	0	40	0.012	0	0.010
◱ 37	0	50	0.012	0	0
◻ 37	0	50	0.012	0	0.010



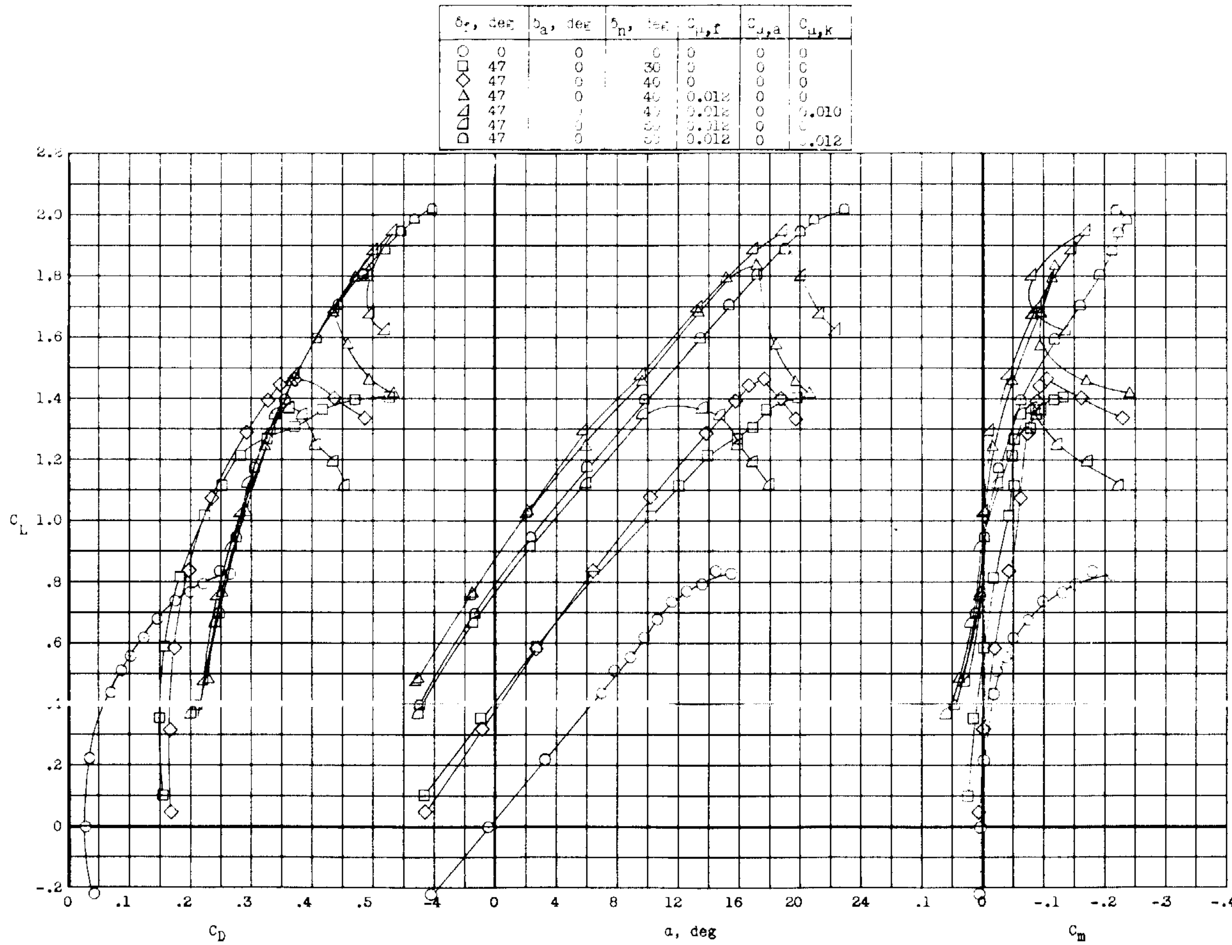
(a) Half-span flaps deflected 37°.

Figure 9.- Comparison of the effects of wing leading-edge droop on the longitudinal characteristics with either half- or full-span flaps deflected 37° to 60°, $i_t = 0°$. With and without boundary-layer control. $z/\bar{c} = -0.09$.



(b) Full-span flaps deflected 37°.

Figure 9.- Continued.



(c) Half-span flaps deflected 47°.

Figure 9.- Continued.

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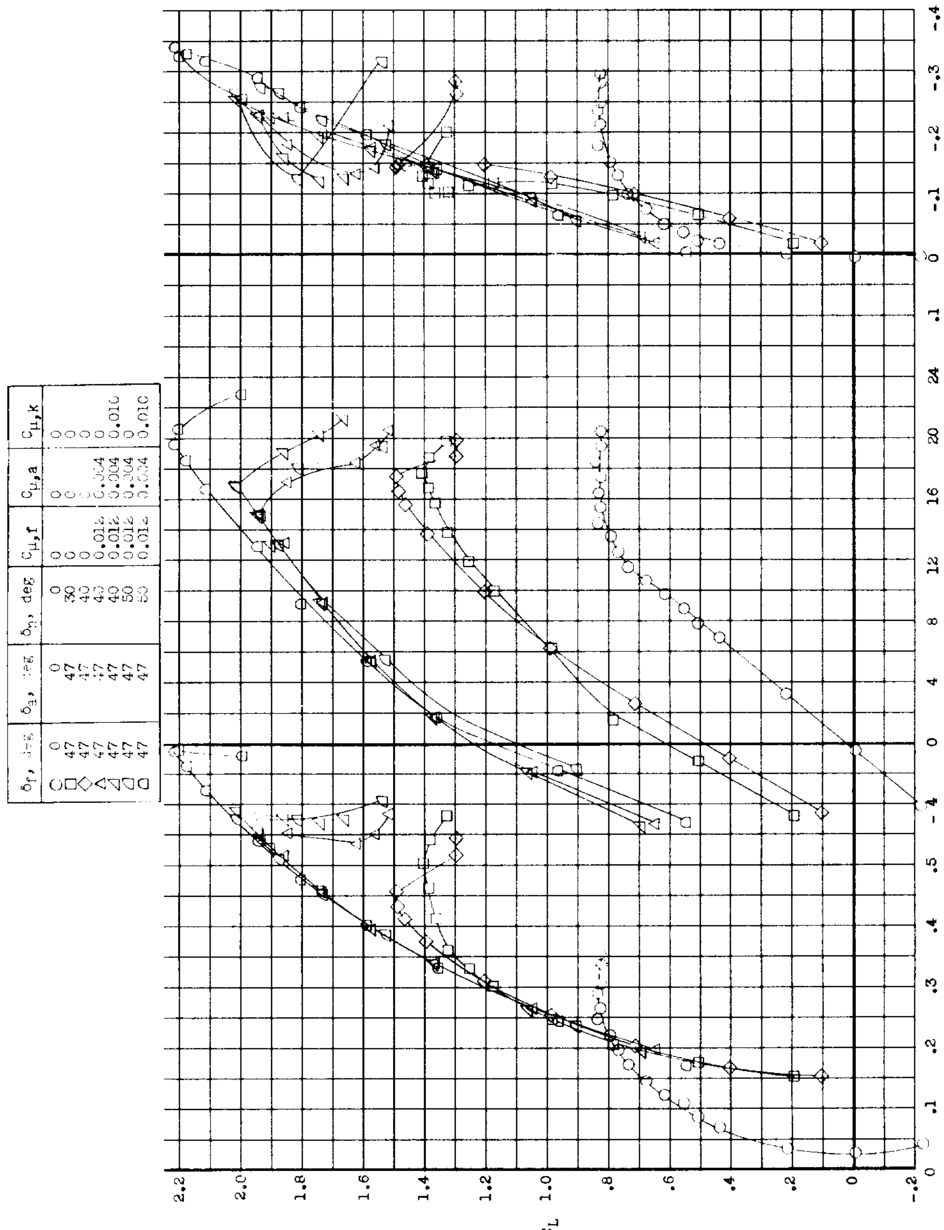
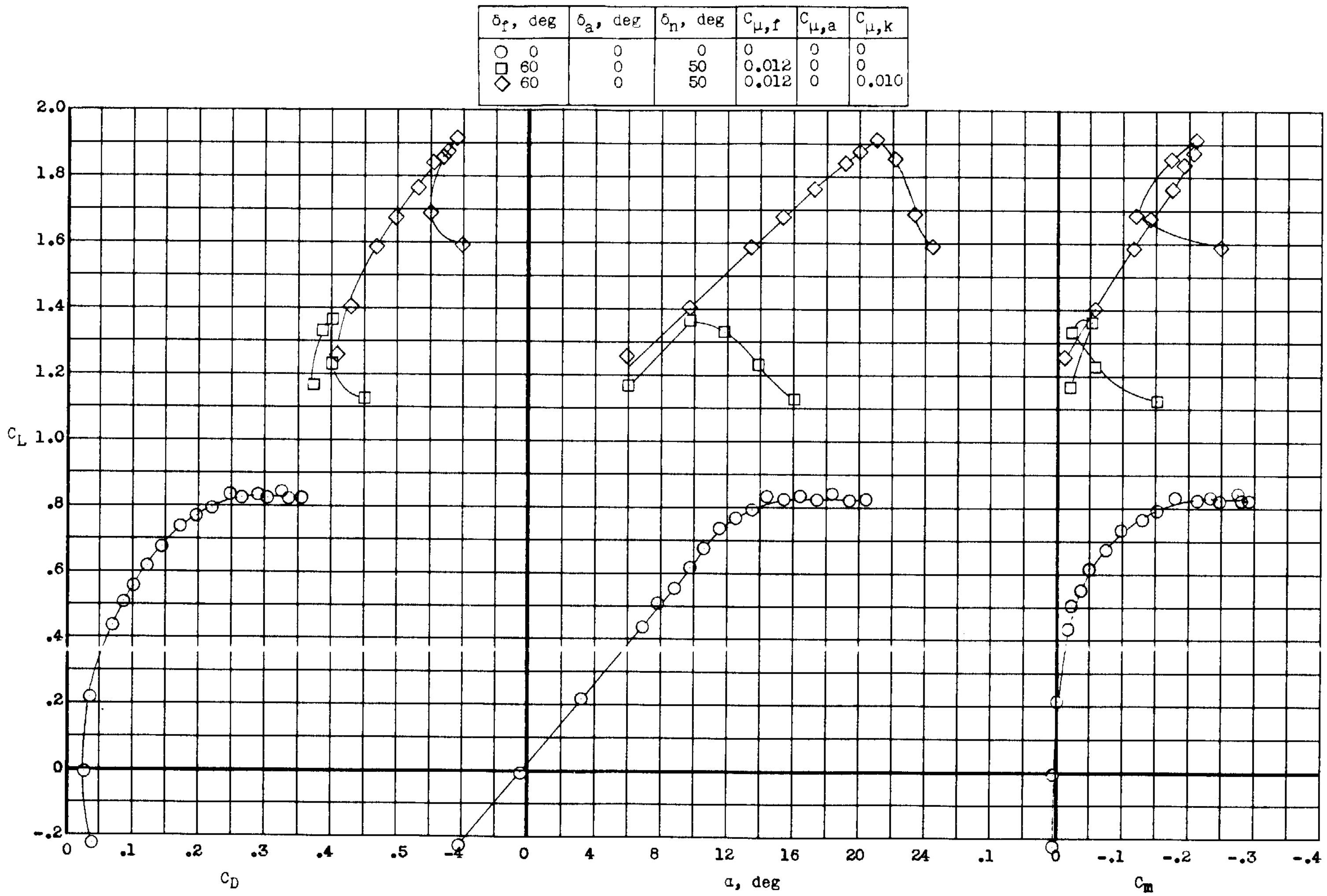


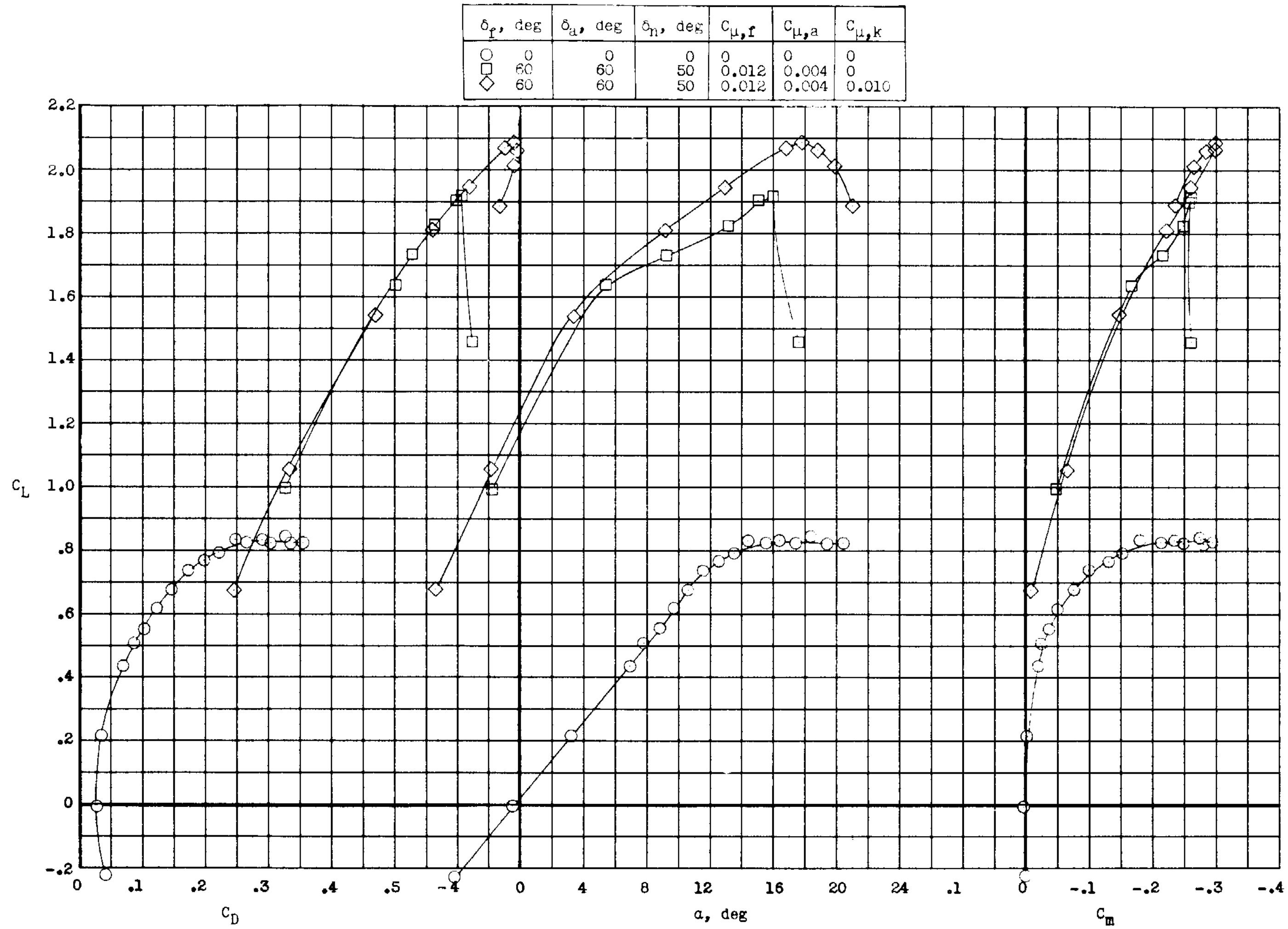
(d) Full-span flaps deflected 47°.

Figure 9.- Continued.



(e) Half-span flaps deflected 60°.

Figure 9.- Continued.



(f) Full-span flaps deflected 60°.

Figure 9.- Concluded.

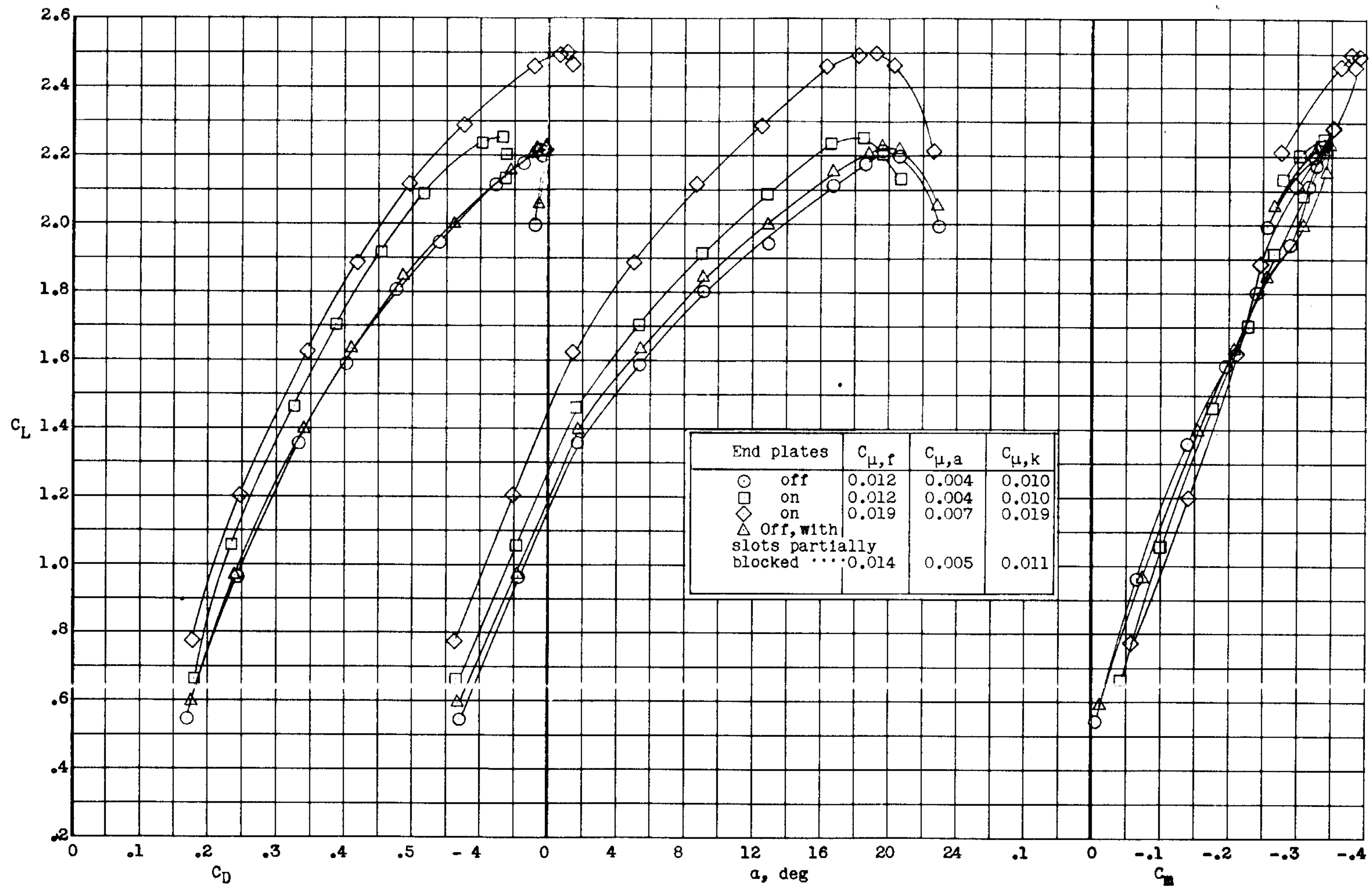
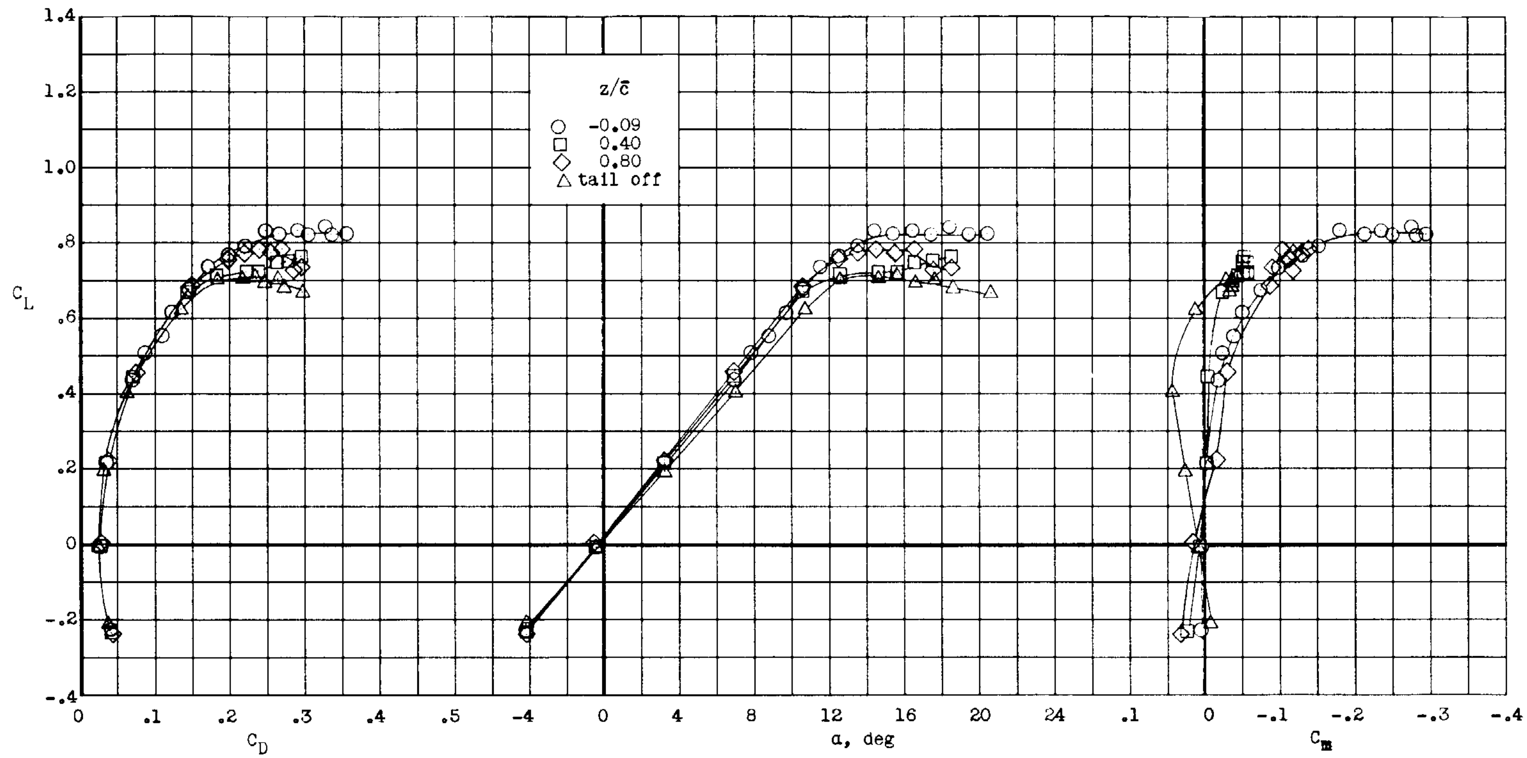
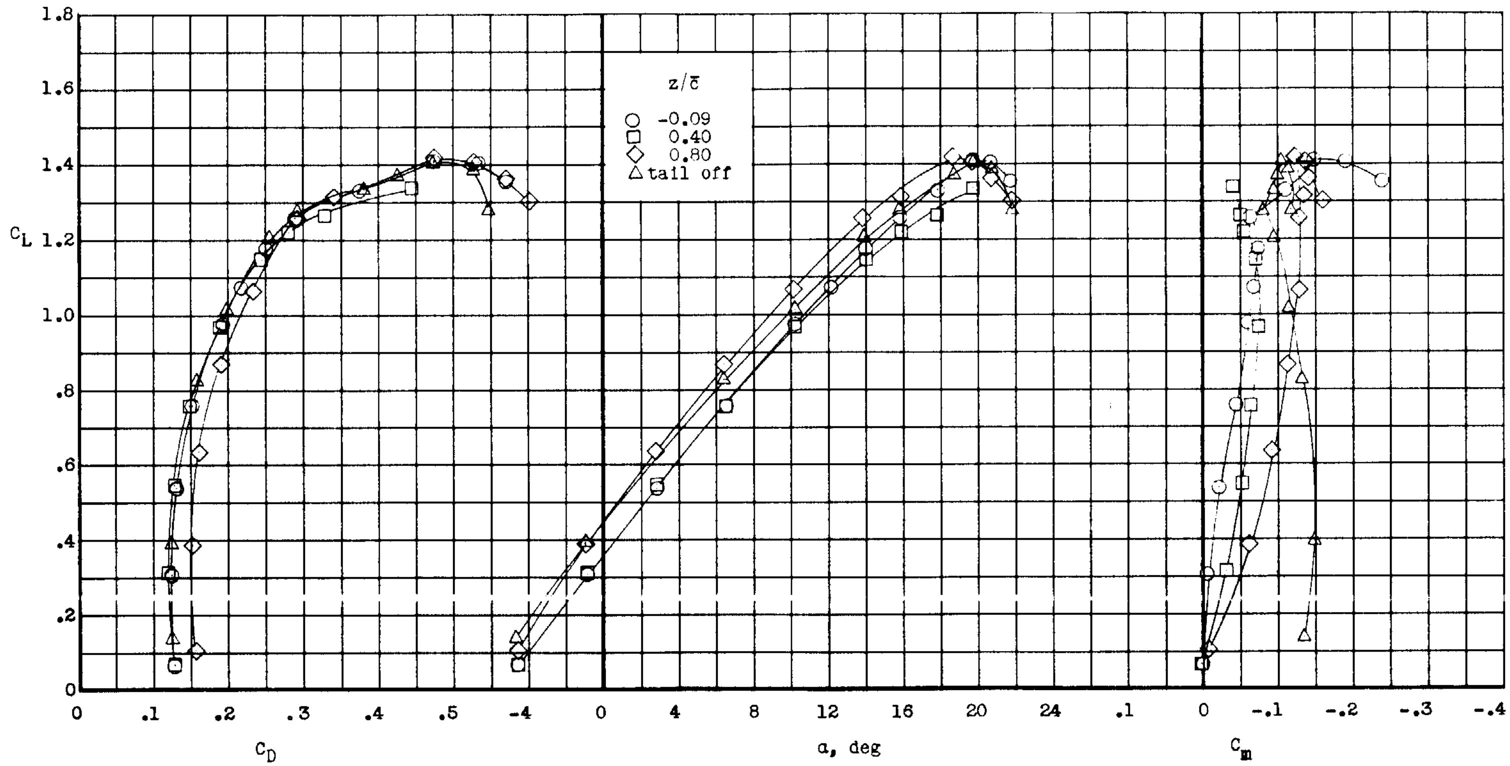


Figure 10.- Effect on the longitudinal characteristics of installing end plates and increasing wing leading-edge and full-span-flap blowing rates. $\delta_f = 47^o$; $\delta_a = 47^o$; $\delta_n = 50^o$; $z/\bar{c} = -0.09$.



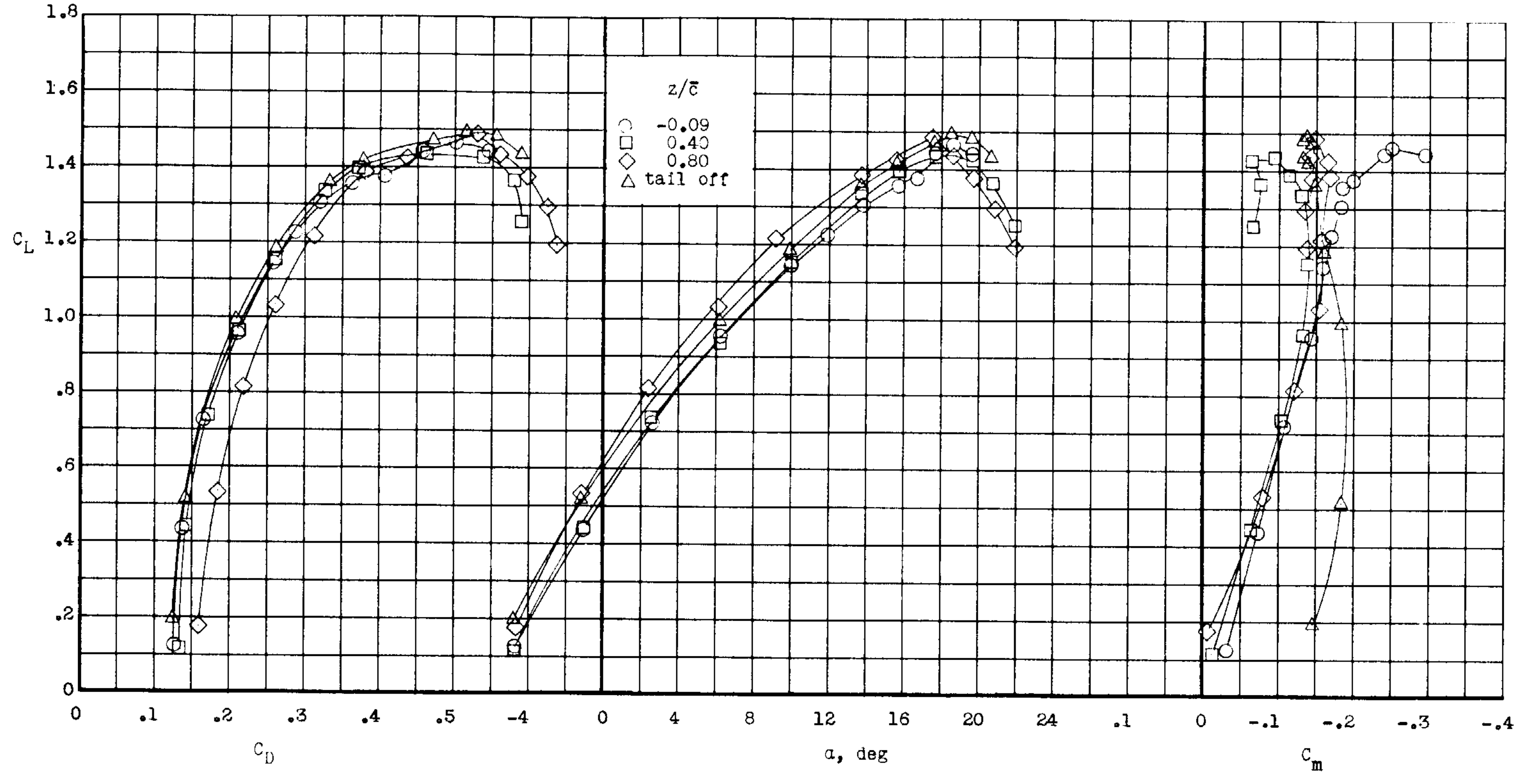
(a) $\delta_f = 0^{\circ}$; $\delta_a = 0^{\circ}$; $\delta_n = 0^{\circ}$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 11.- Effect on the longitudinal characteristics of varying the horizontal-tail height for several wing configurations with and without boundary-layer control. $i_t = 0^{\circ}$.



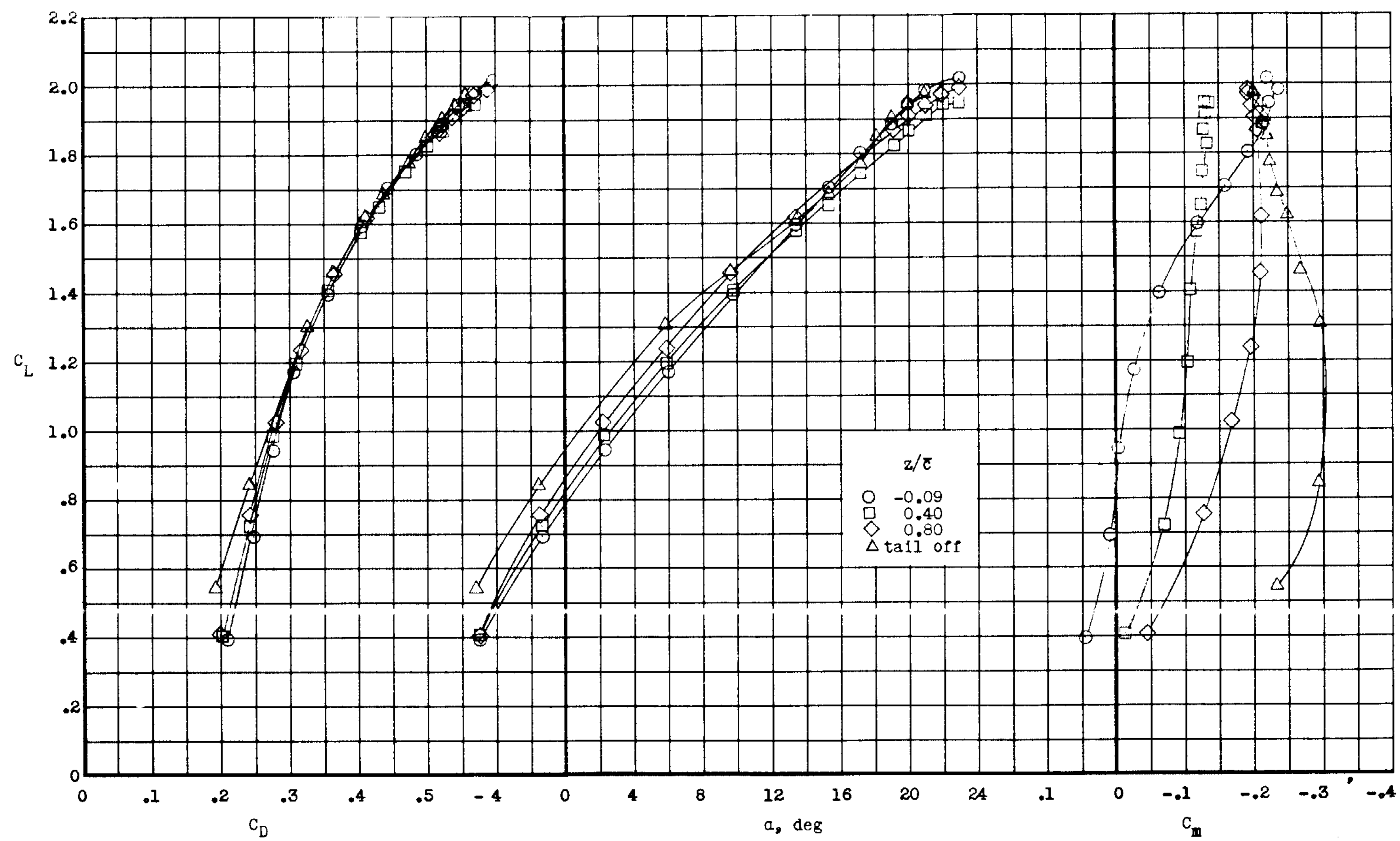
(b) $\delta_f = 37^o$; $\delta_a = 0^o$; $\delta_n = 30^o$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 11.- Continued.



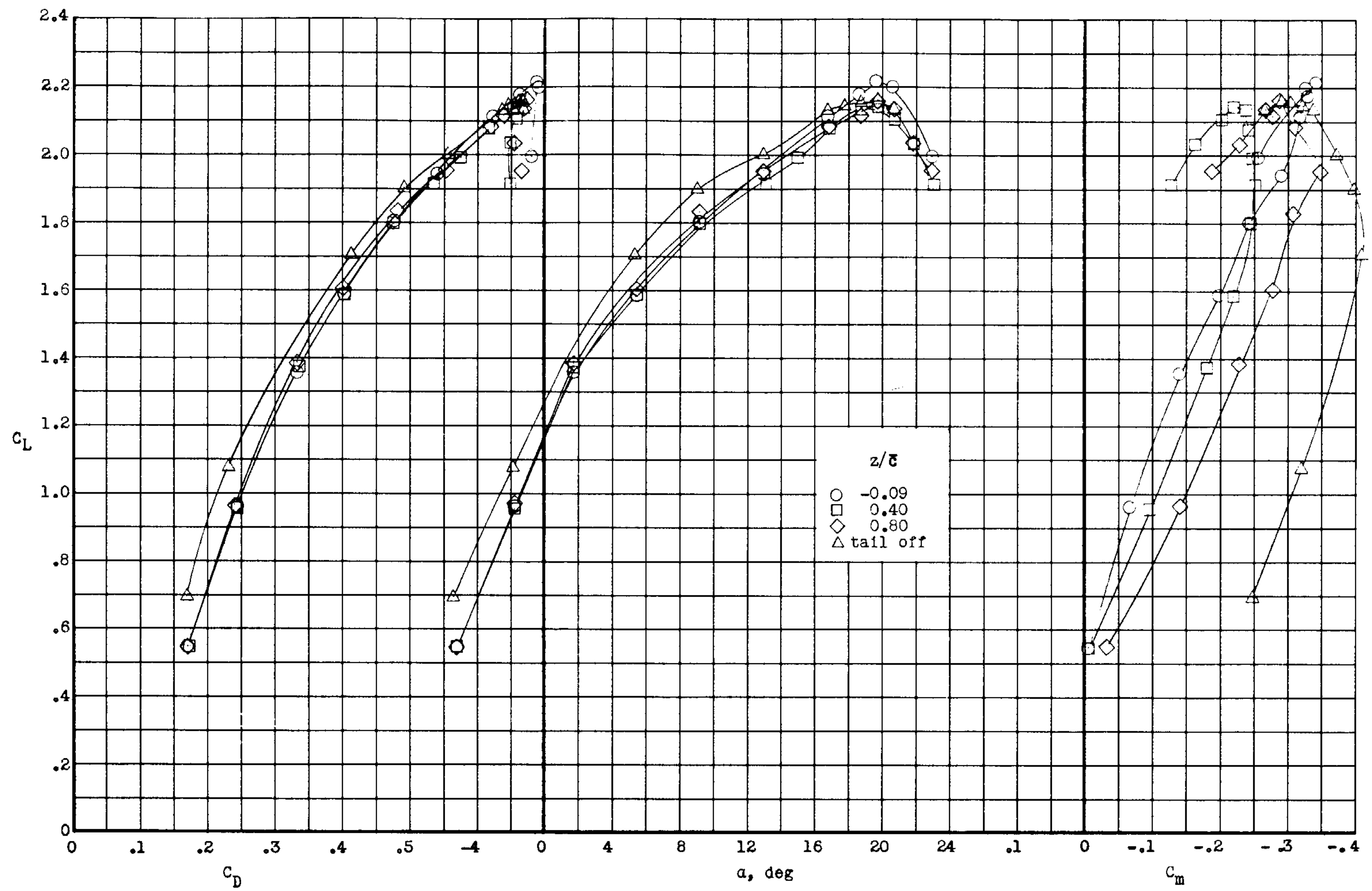
(c) $\delta_f = 37^\circ$; $\delta_a = 37^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 11.- Continued.



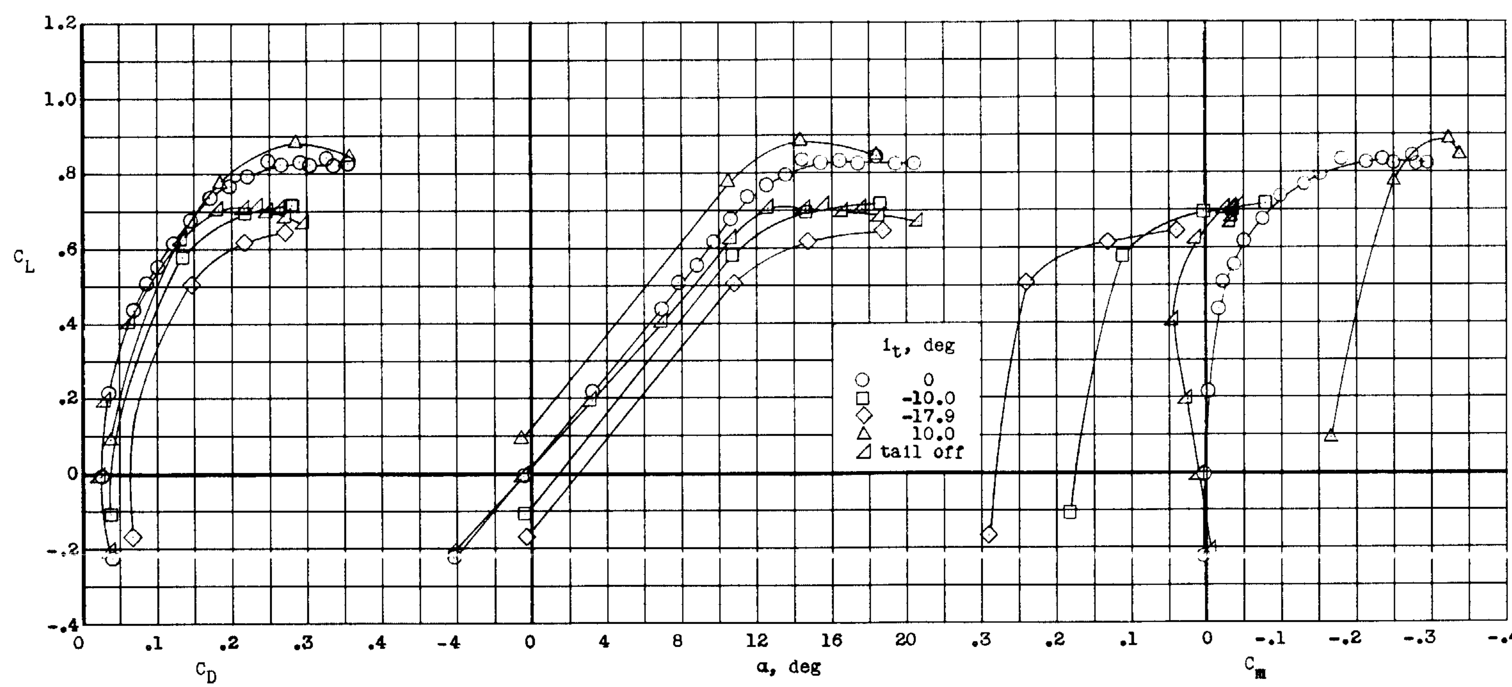
(d) $\delta_f = 47^{\circ}$; $\delta_a = 0^{\circ}$; $\delta_n = 50^{\circ}$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 11.- Continued.



(e) $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 11.- Concluded.



(a) $\delta_f = 0^\circ$; $\delta_a = 0^\circ$; $\delta_n = 0^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 12.- Effect of horizontal-tail deflection on the longitudinal characteristics of several wing configurations with and without boundary-layer control. $z/\bar{c} = -0.09$.

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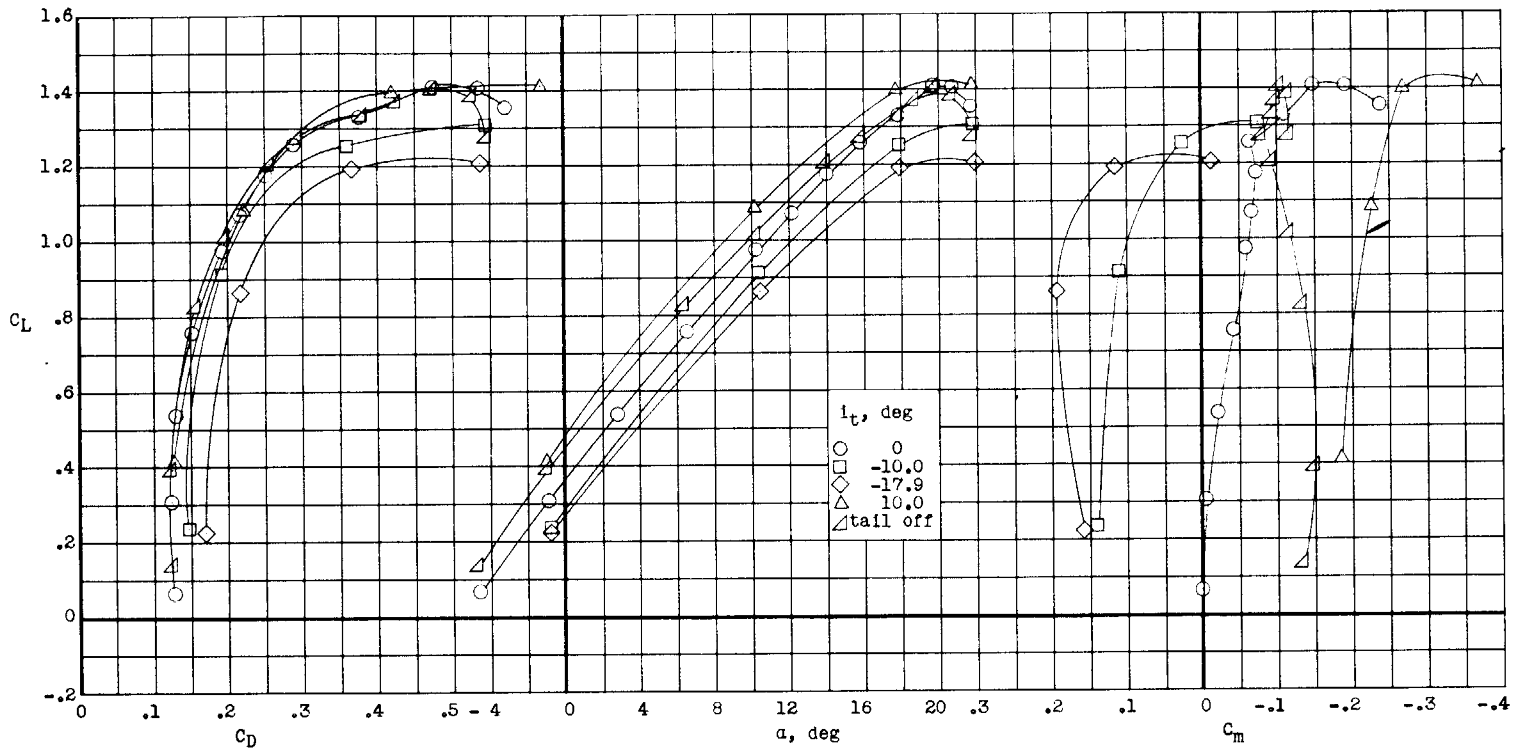
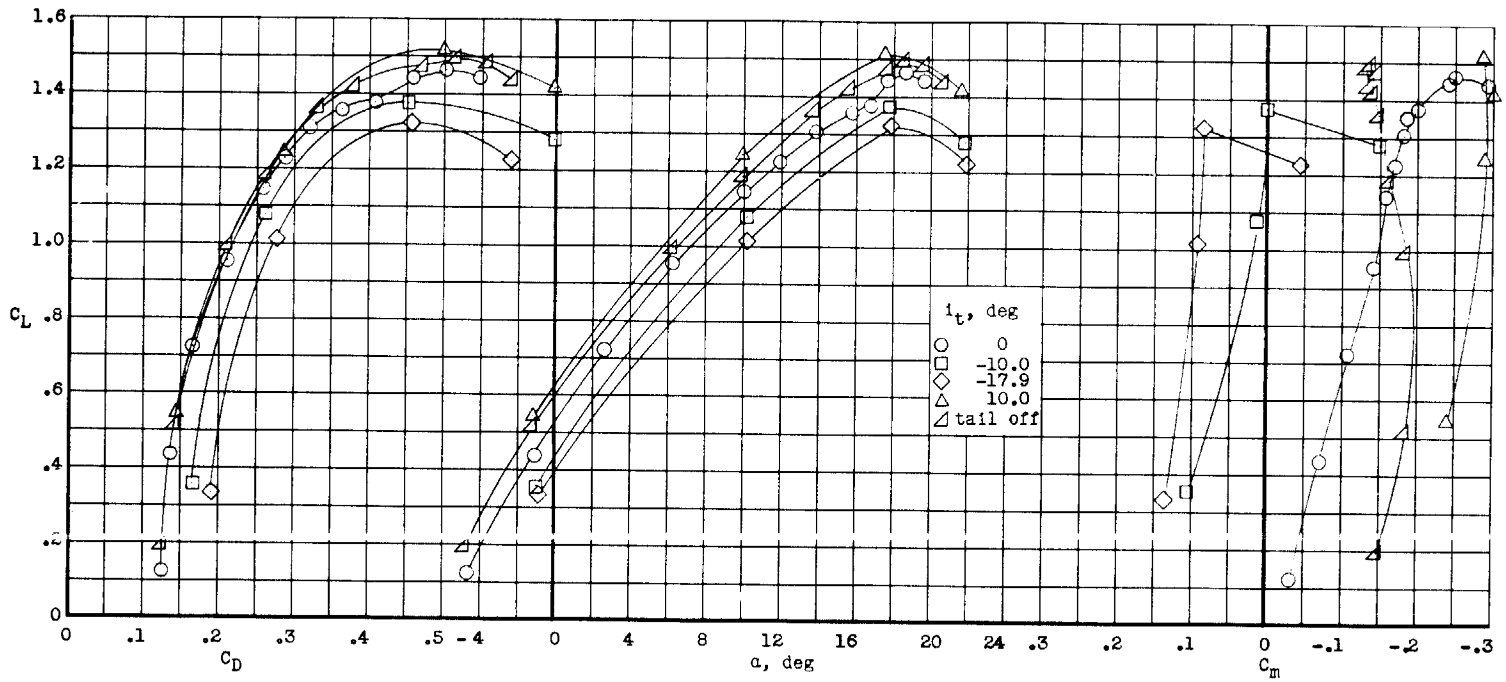


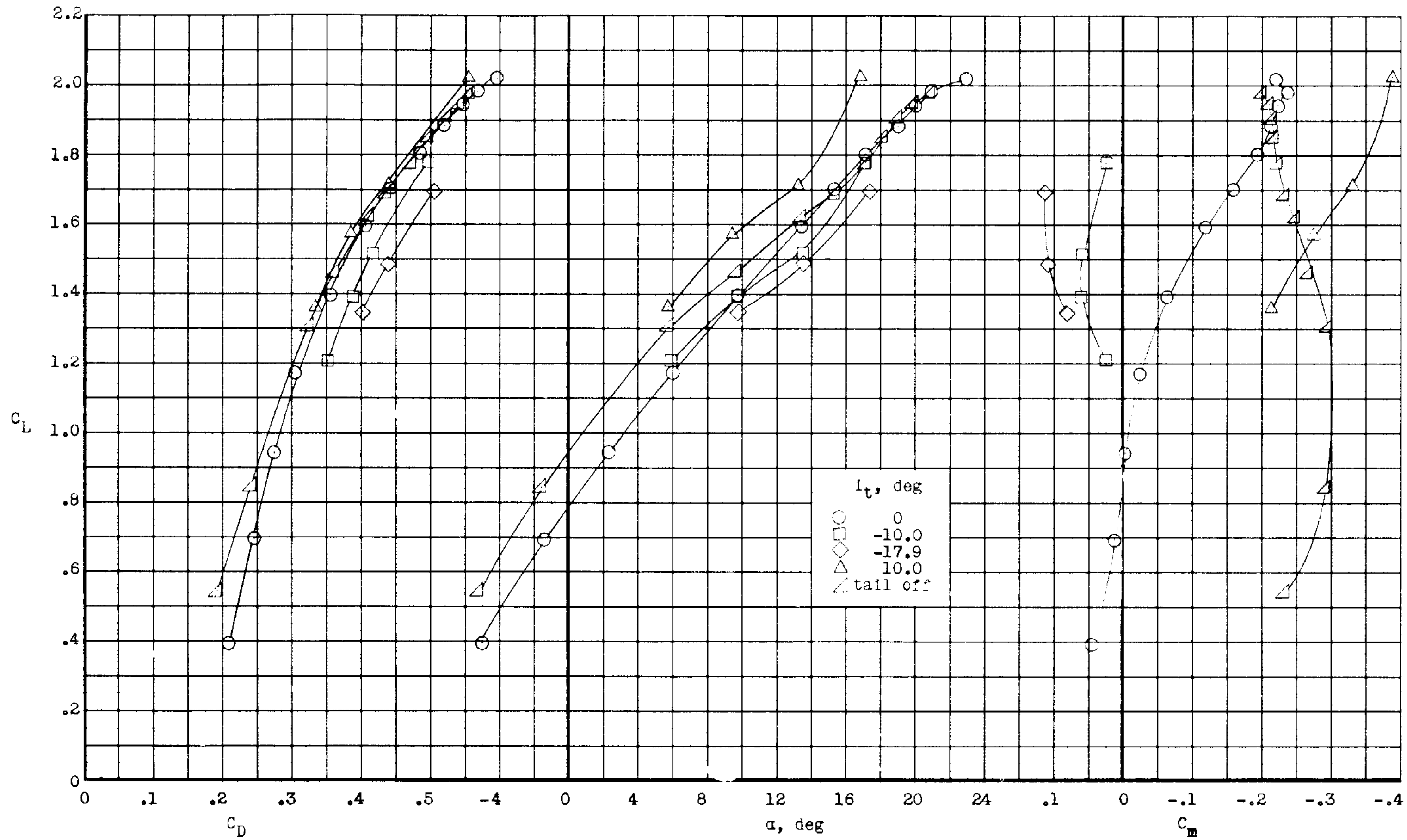
(b) $\delta_f = 37^{\circ}$; $\delta_a = 0^{\circ}$; $\delta_n = 30^{\circ}$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 12.- Continued.



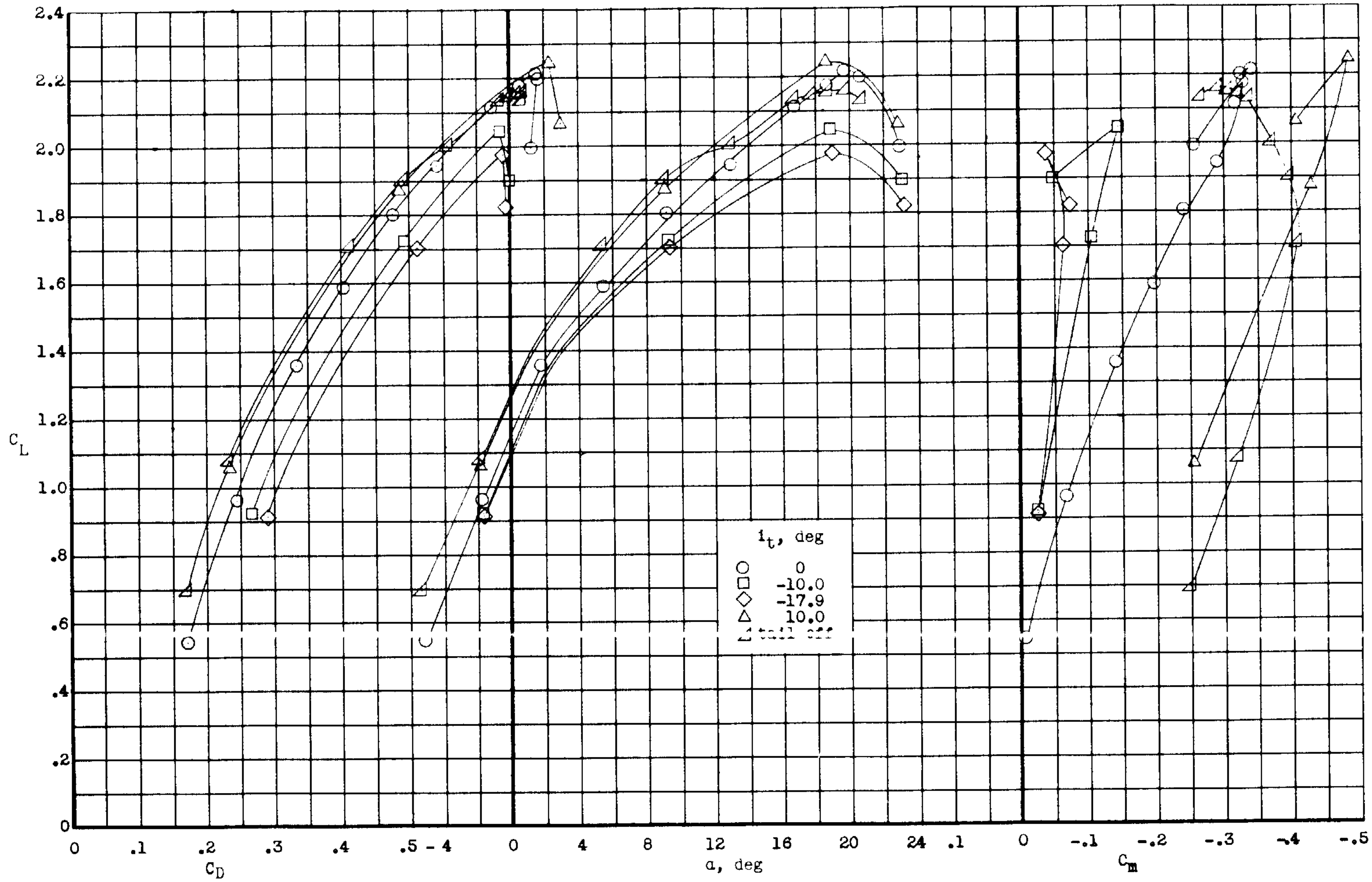
(c) $\delta_f = 37^o$; $\delta_a = 37^o$; $\delta_n = 30^o$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 12.- Continued.



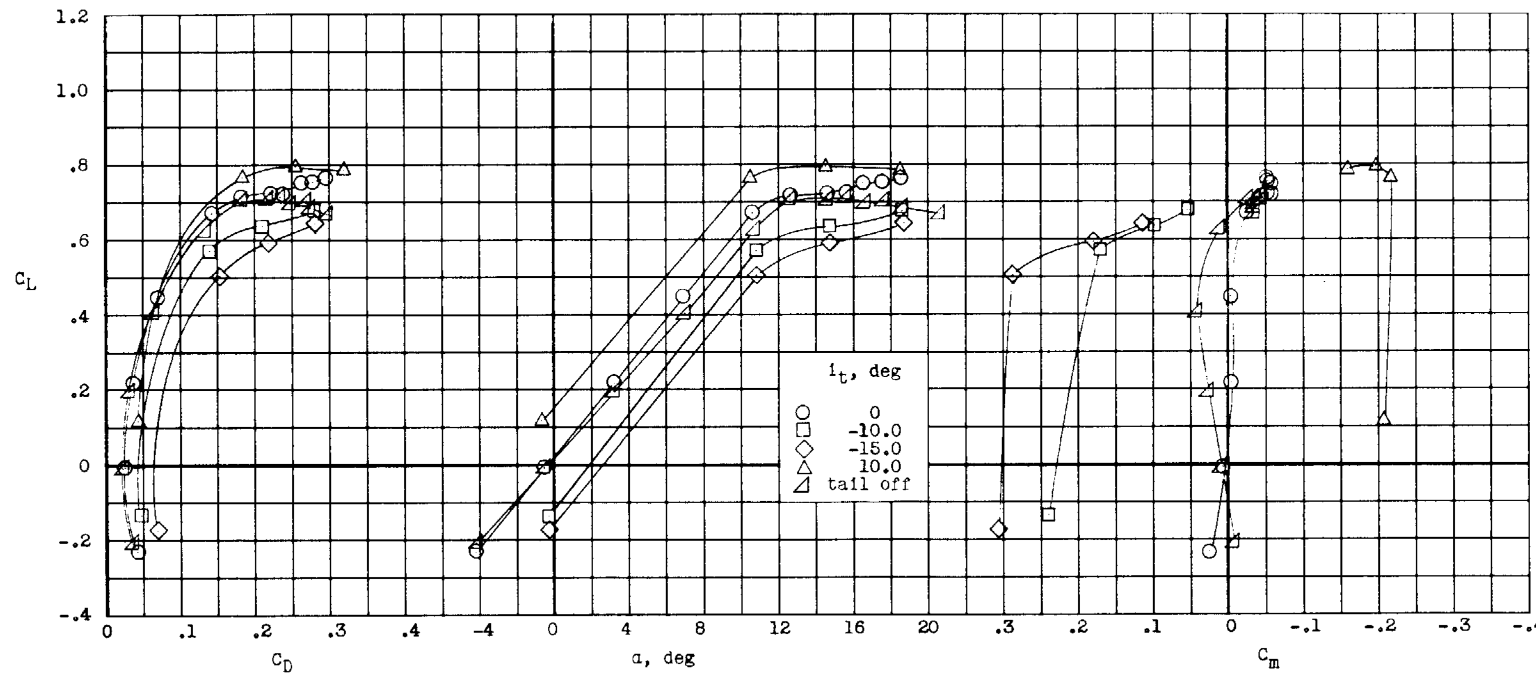
(d) $\delta_f = 47^\circ$; $\delta_a = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 12.- Continued.



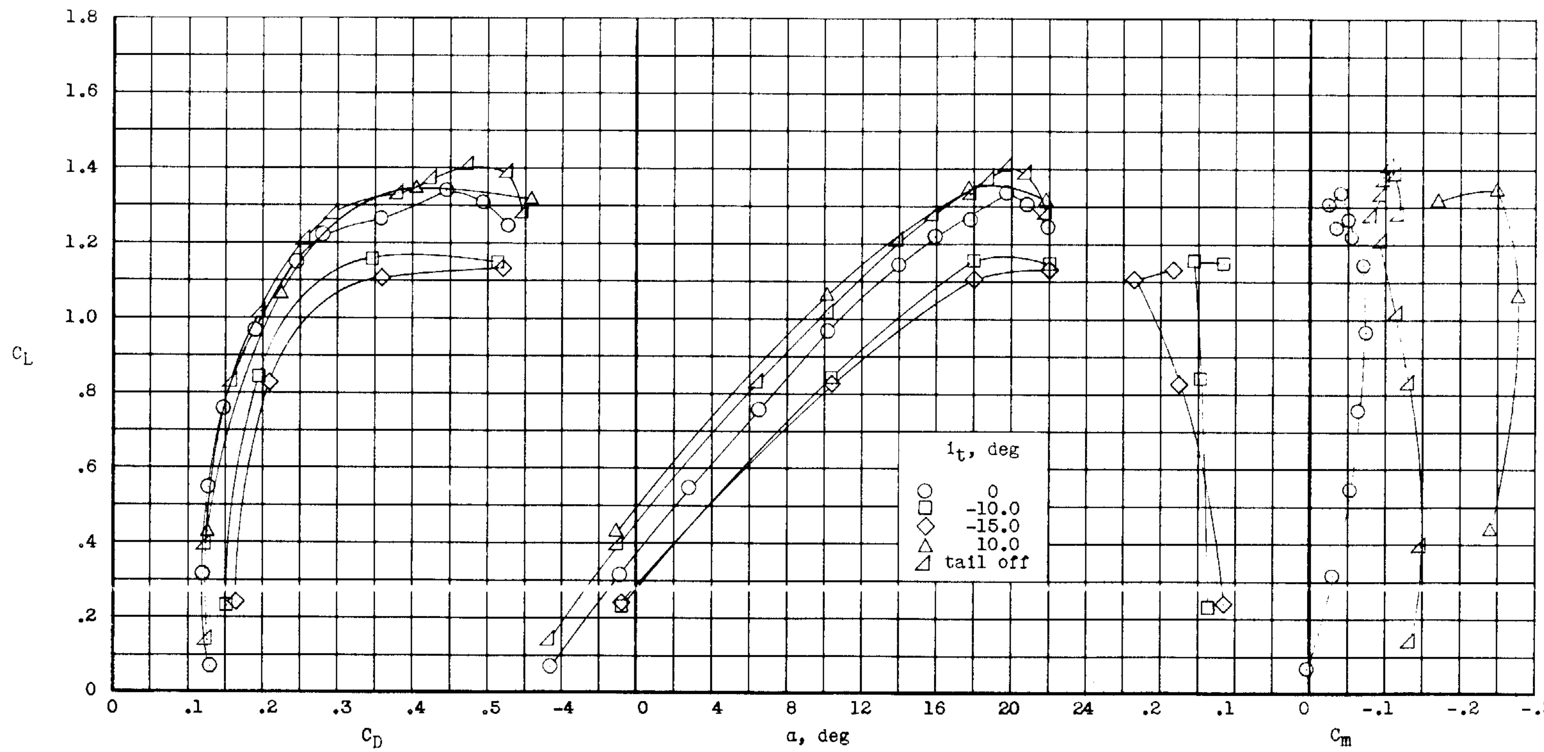
(e) $\delta_f = 47^o$; $\delta_a = 47^o$; $\delta_n = 50^o$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 12.- Concluded.



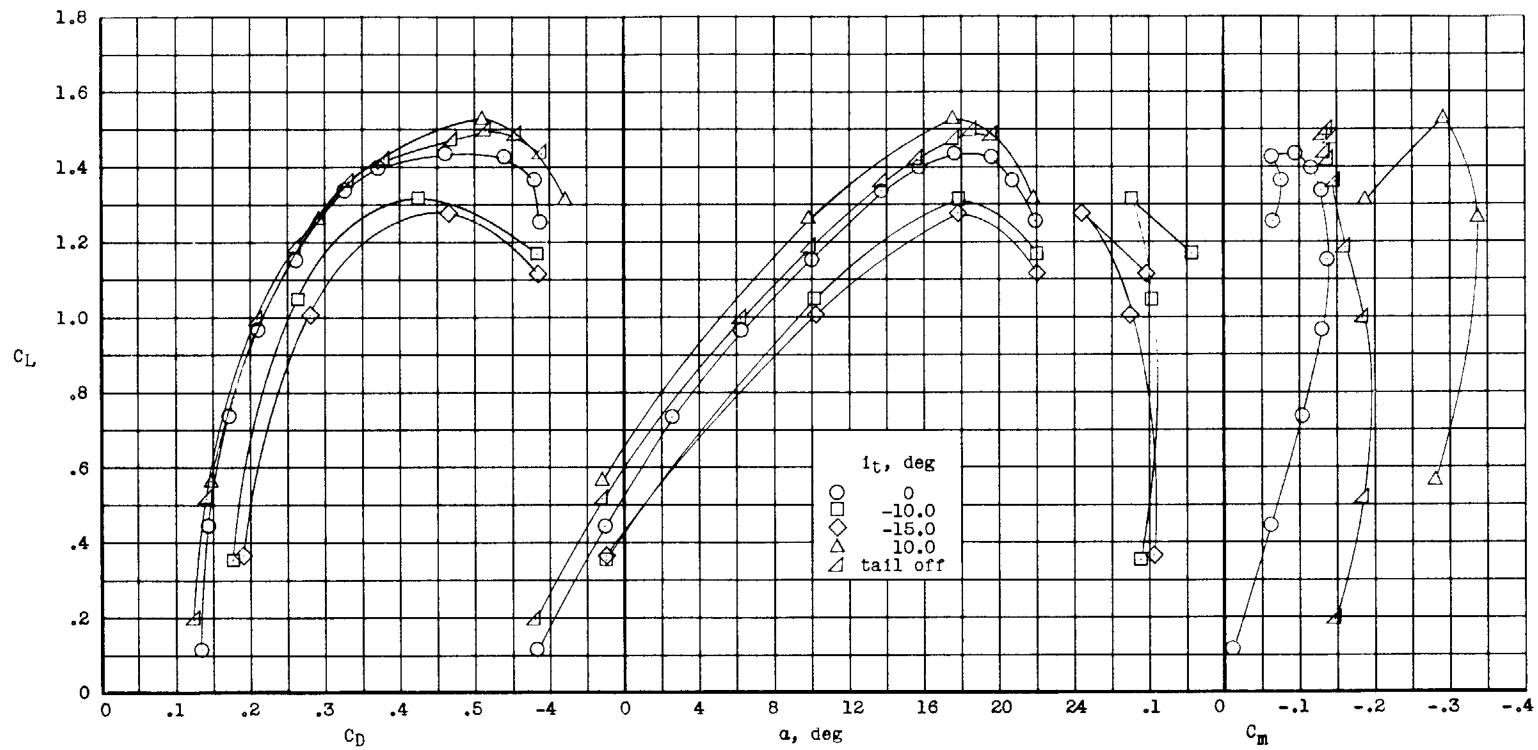
(a) $\delta_f = 0^\circ$; $\delta_a = 0^\circ$; $\delta_n = 0^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 13.- Effect of horizontal-tail deflection on the longitudinal characteristics of several wing configurations with and without boundary-layer control. $z/\bar{c} = 0.40$.



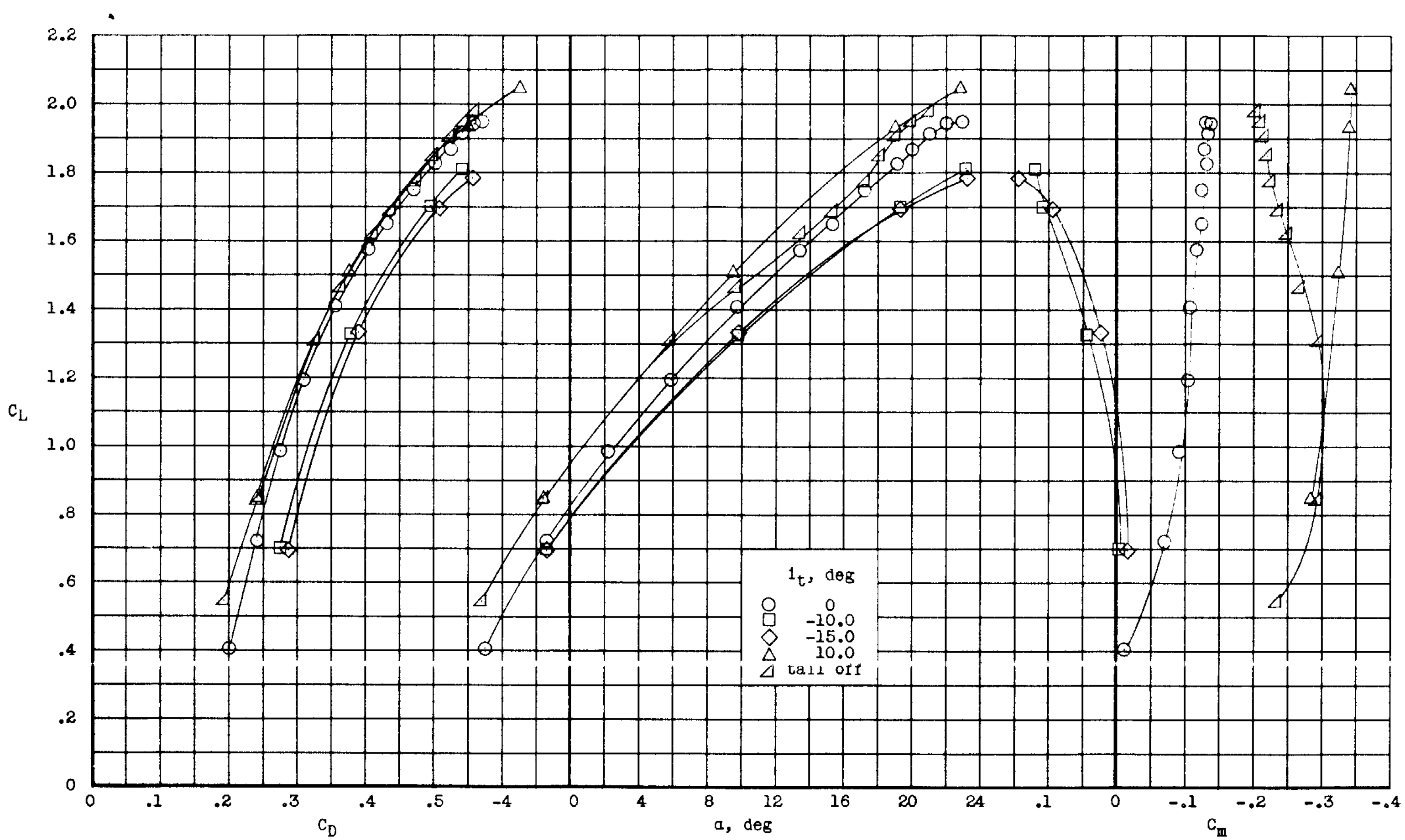
(b) $\delta_f = 37^{\circ}$; $\delta_a = 0^{\circ}$; $\delta_n = 30$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 13.- Continued.



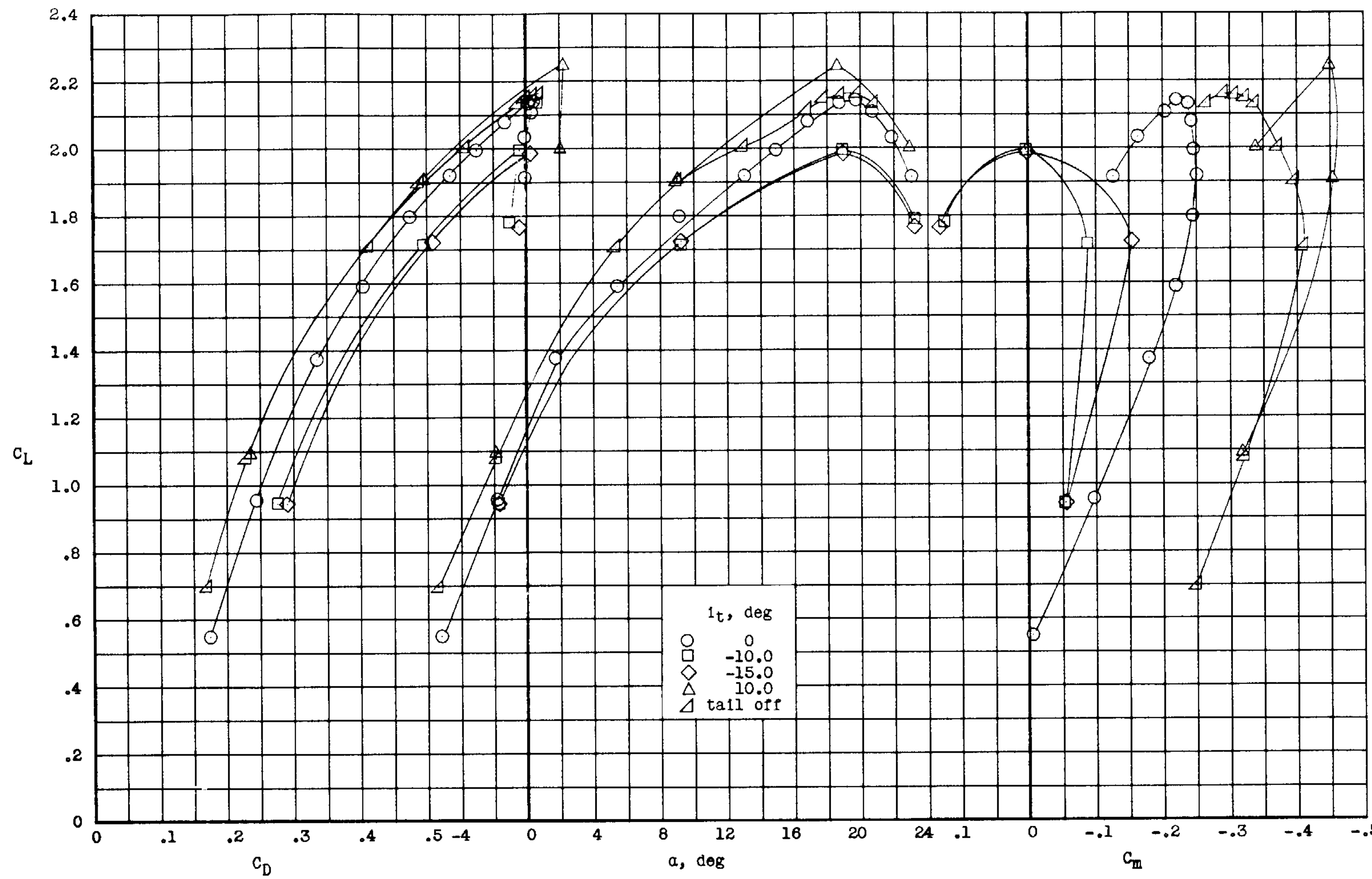
(c) $\delta_f = 37^\circ$; $\delta_a = 37^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 13.- Continued.



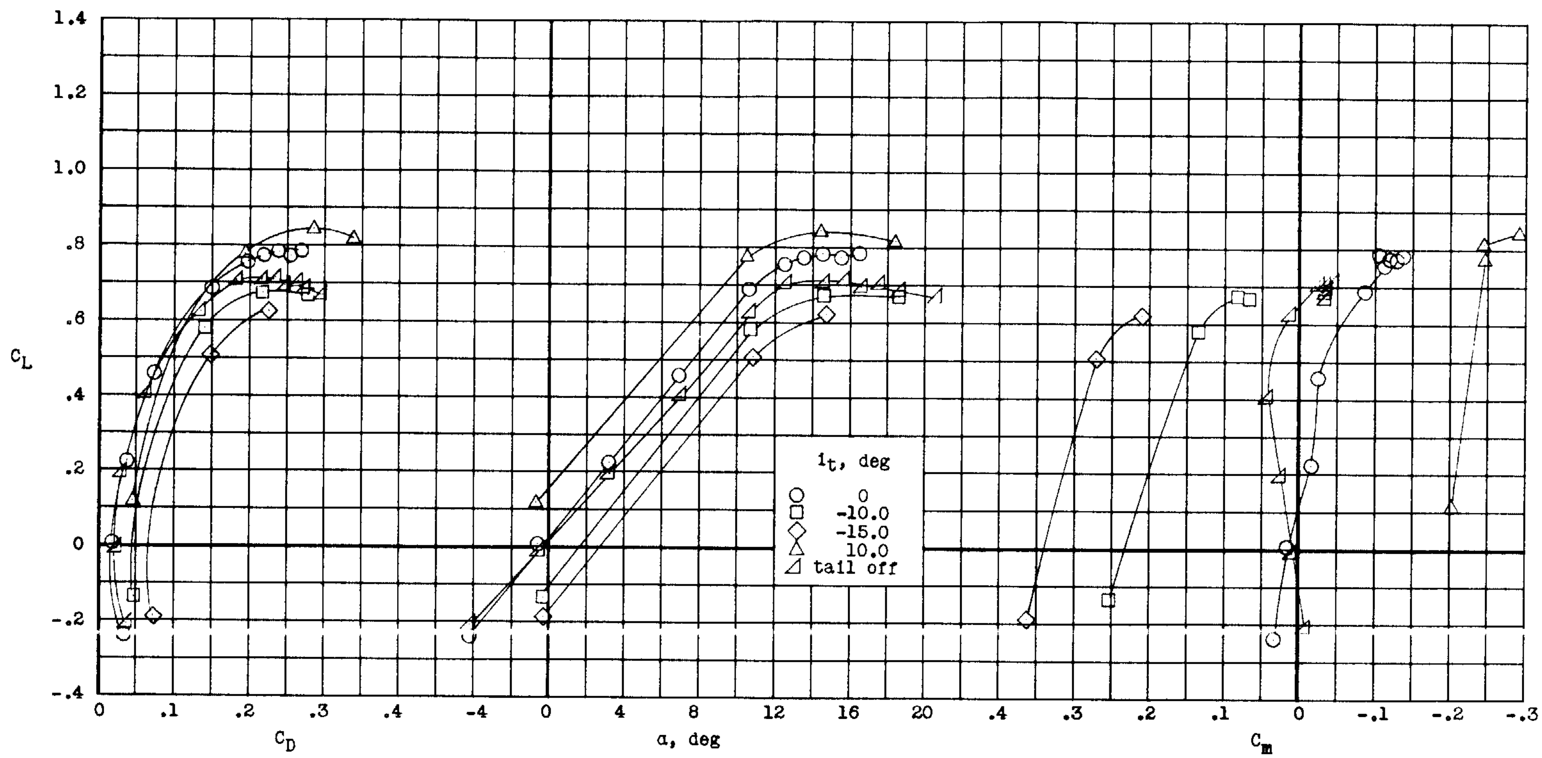
(d) $\delta_f = 47^\circ$; $\delta_a = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 13.- Continued.



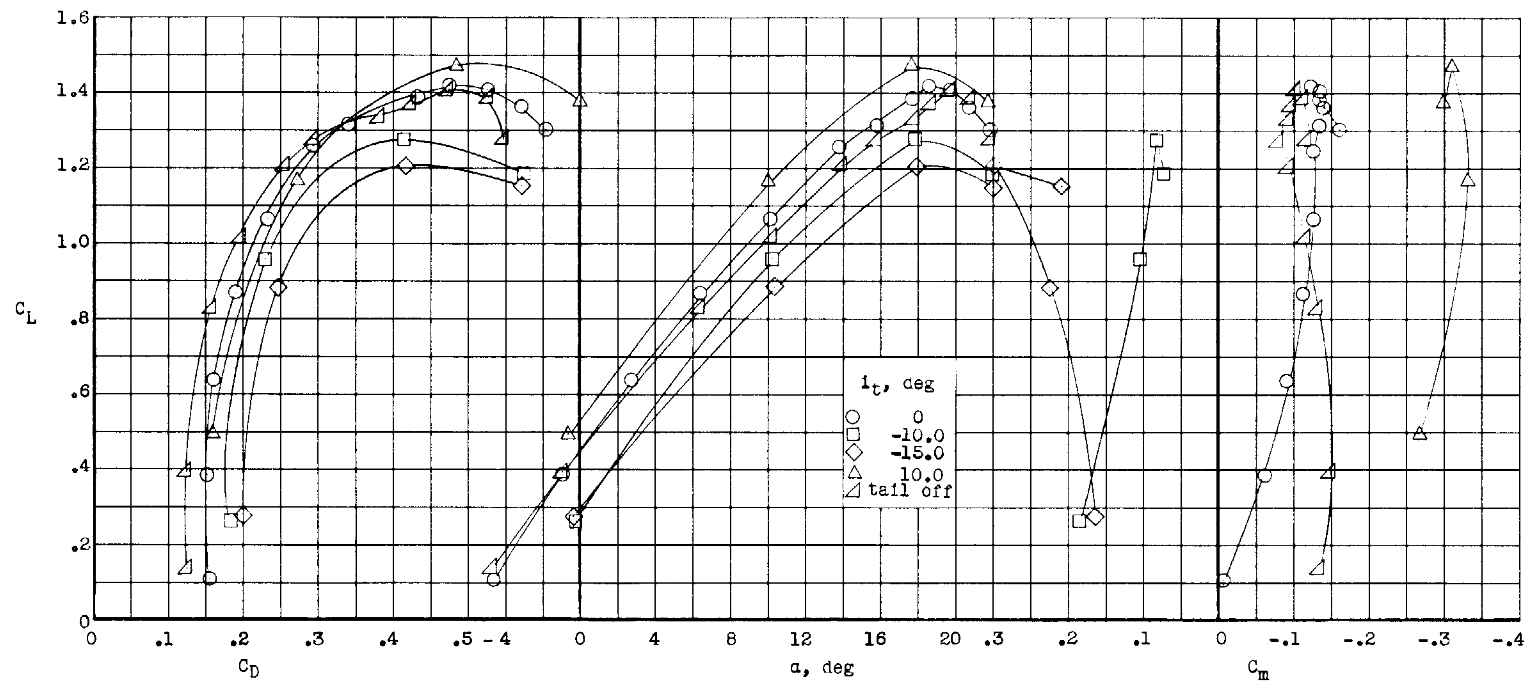
(e) $\delta_f = 47^o$; $\delta_a = 47^o$; $\delta_n = 50^o$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 13.- Concluded.



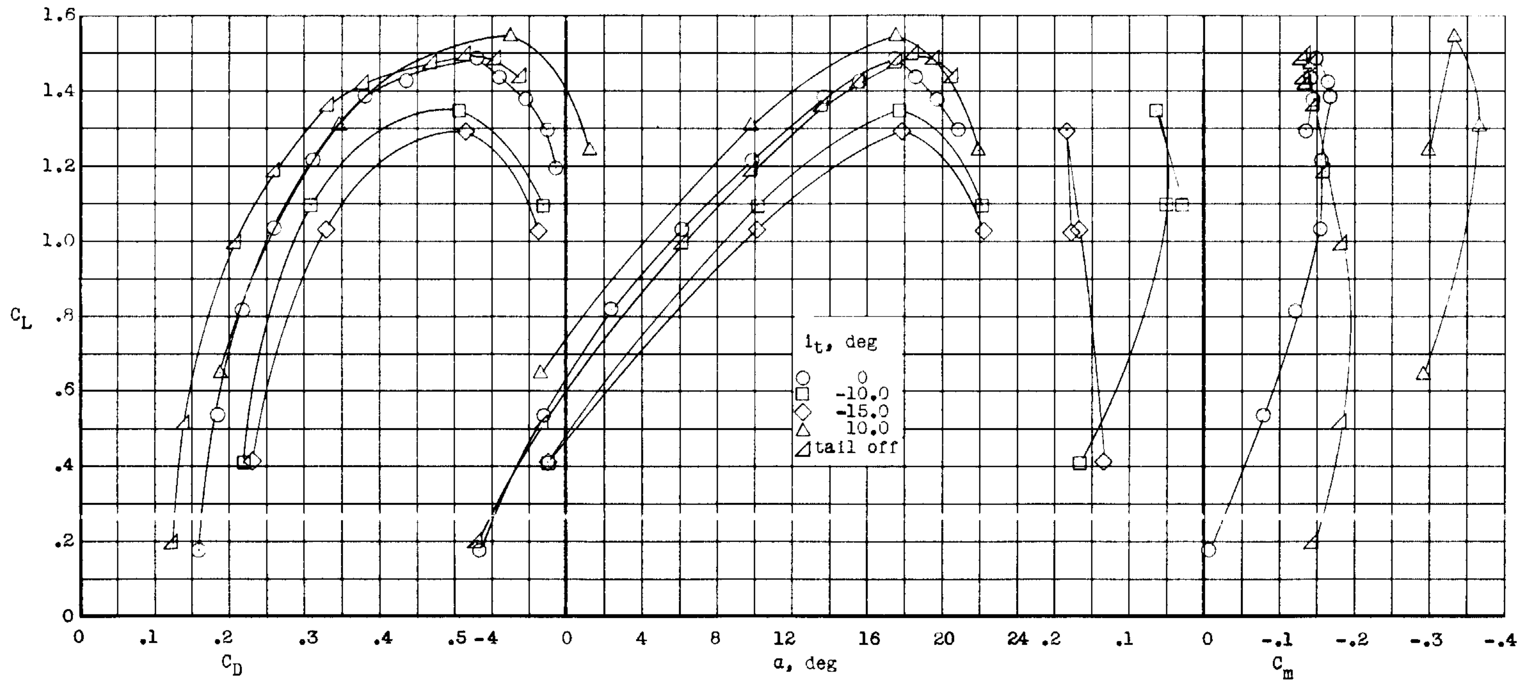
(a) $\delta_f = 0^{\circ}$; $\delta_a = 0^{\circ}$; $\delta_n = 0^{\circ}$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 14.- Effect of horizontal-tail deflection on the longitudinal characteristics of several wing configurations with and without boundary-layer control. $z/\bar{c} = 0.80$.



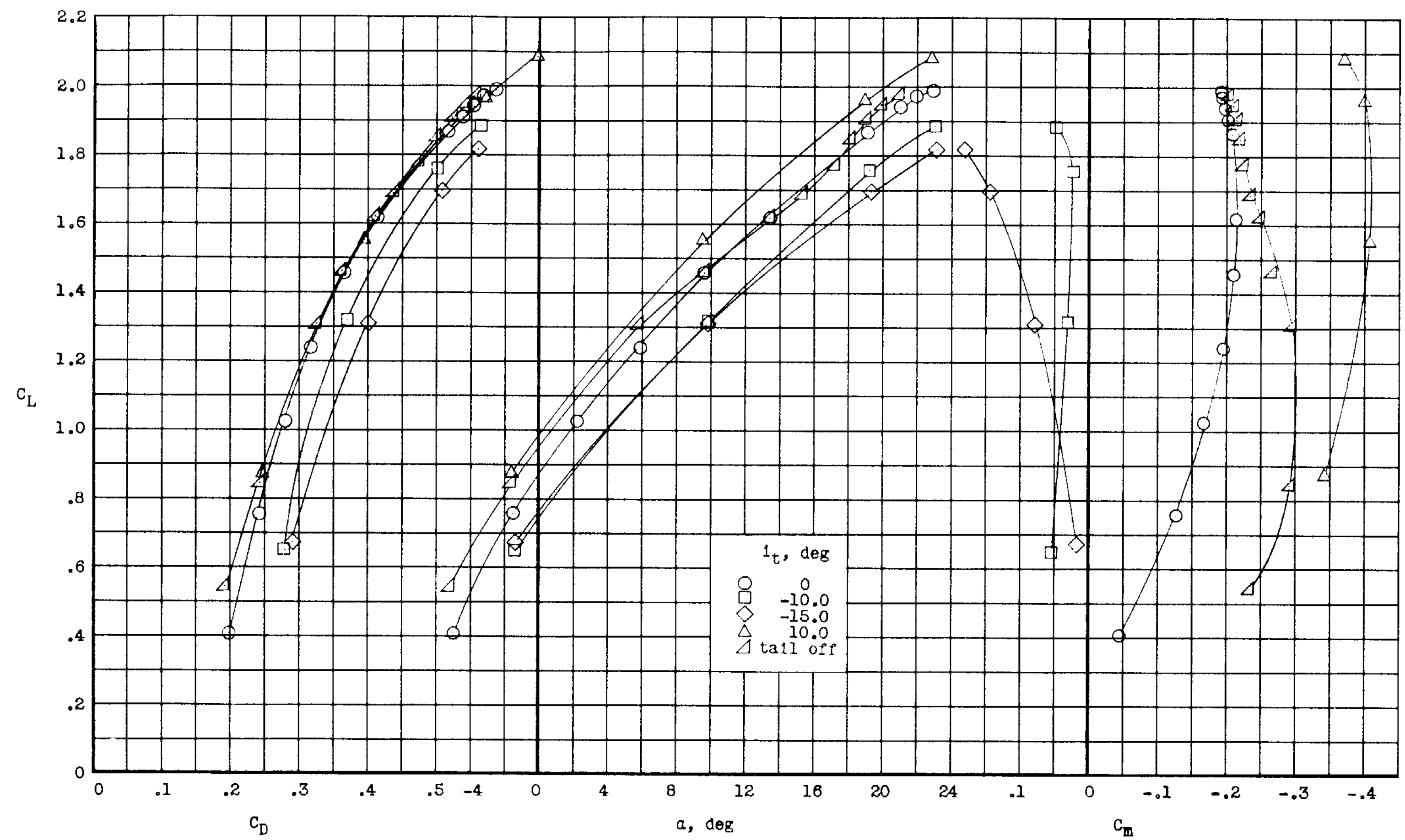
(b) $\delta_f = 37^\circ$; $\delta_a = 0^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 14.- Continued.



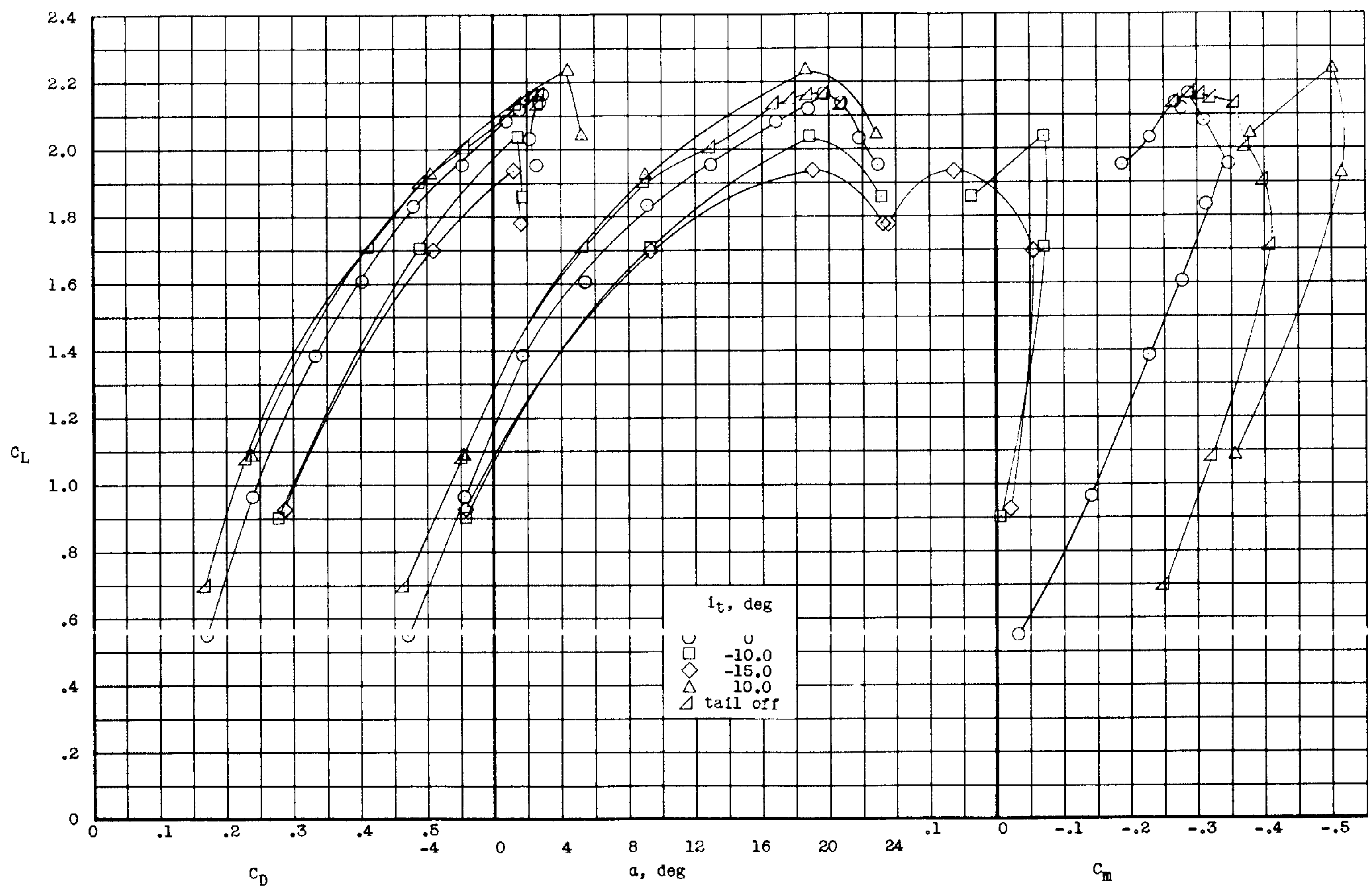
(c) $\delta_f = 37^\circ$; $\delta_a = 37^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 14.- Continued.



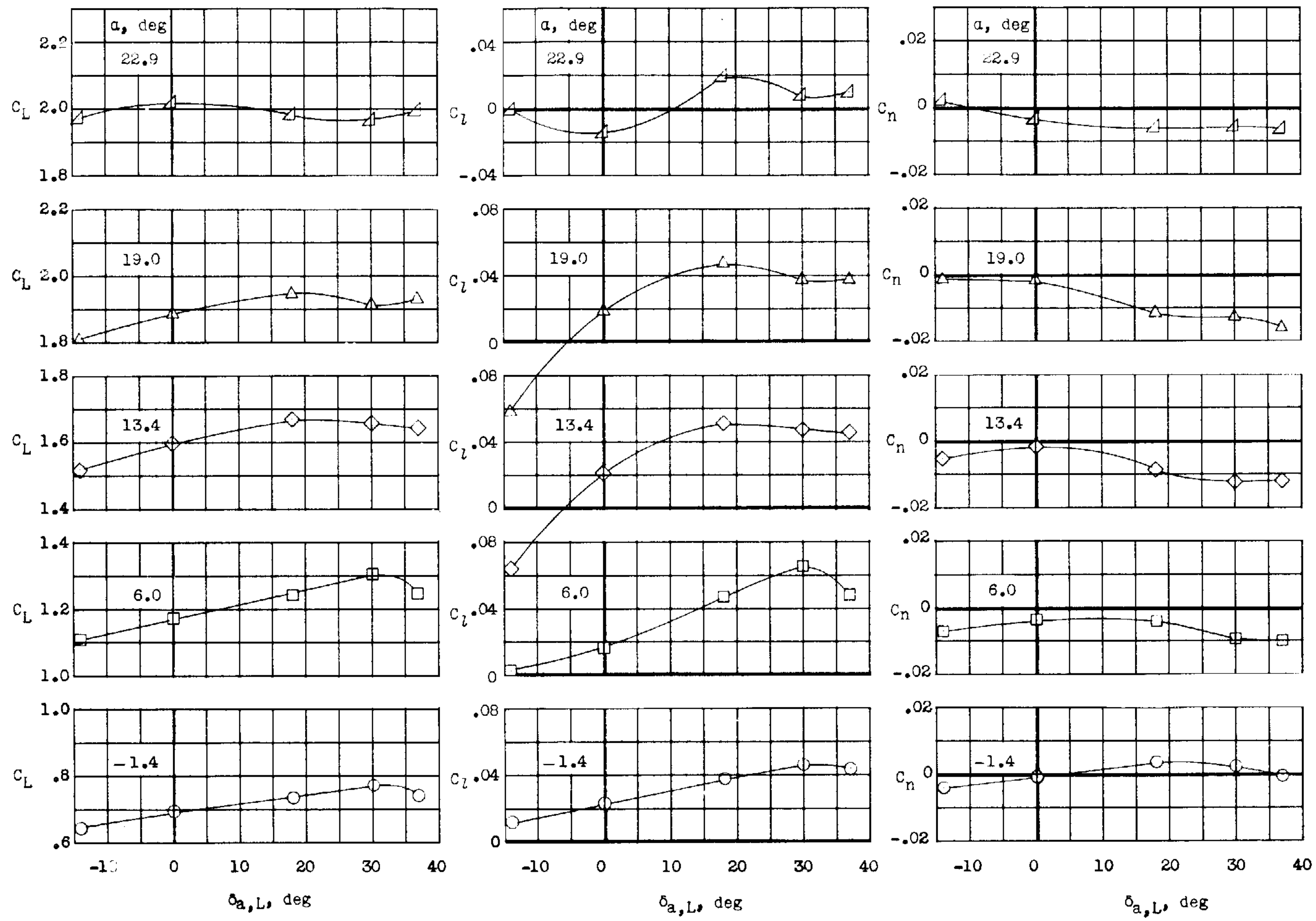
(d) $\delta_f = 47^o$; $\delta_a = 0^o$; $\delta_n = 50^o$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 14.- Continued.



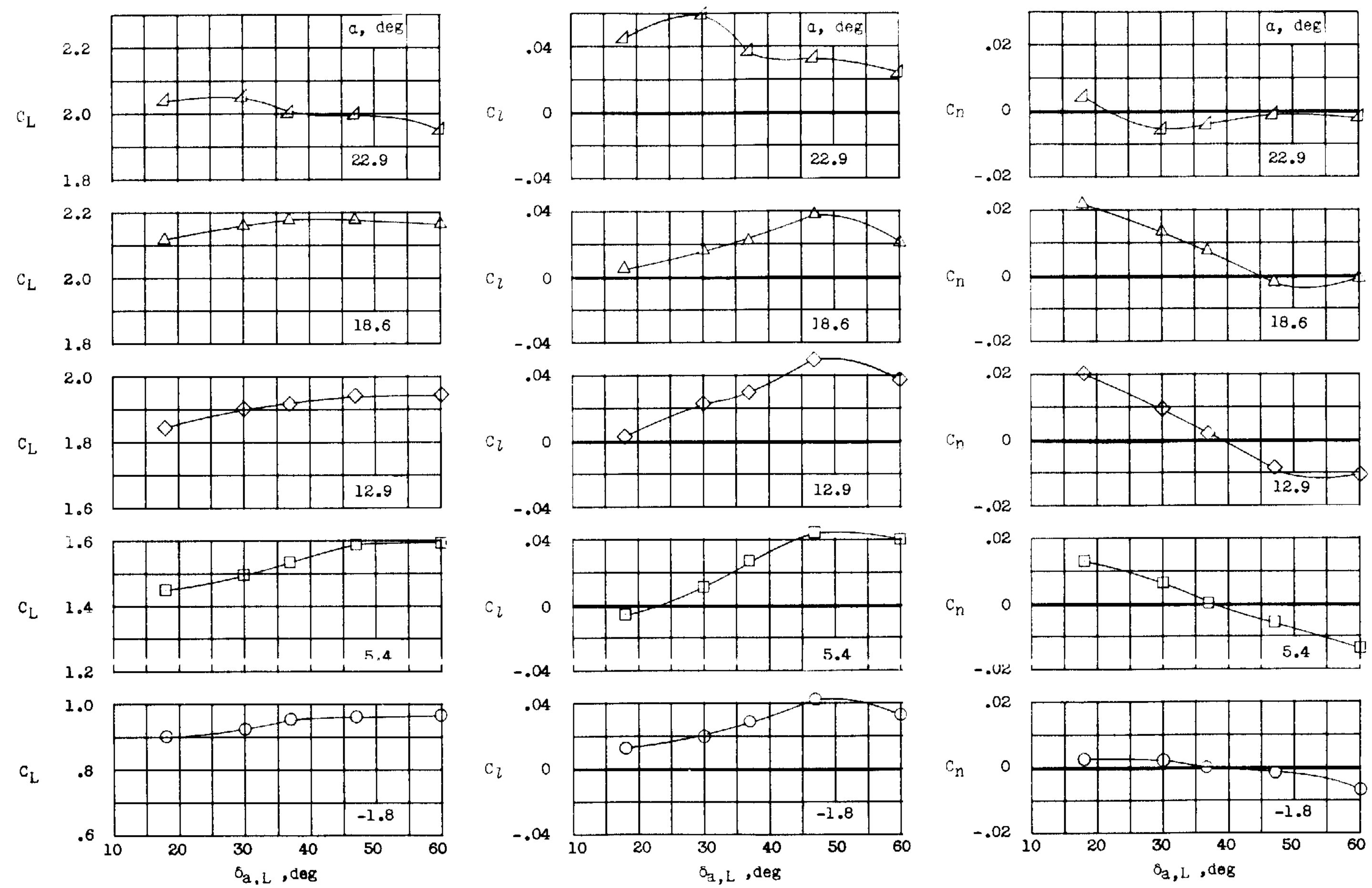
(e) $\delta_f = 47^o$; $\delta_a = 47^o$; $\delta_n = 50^o$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 14.- Concluded.



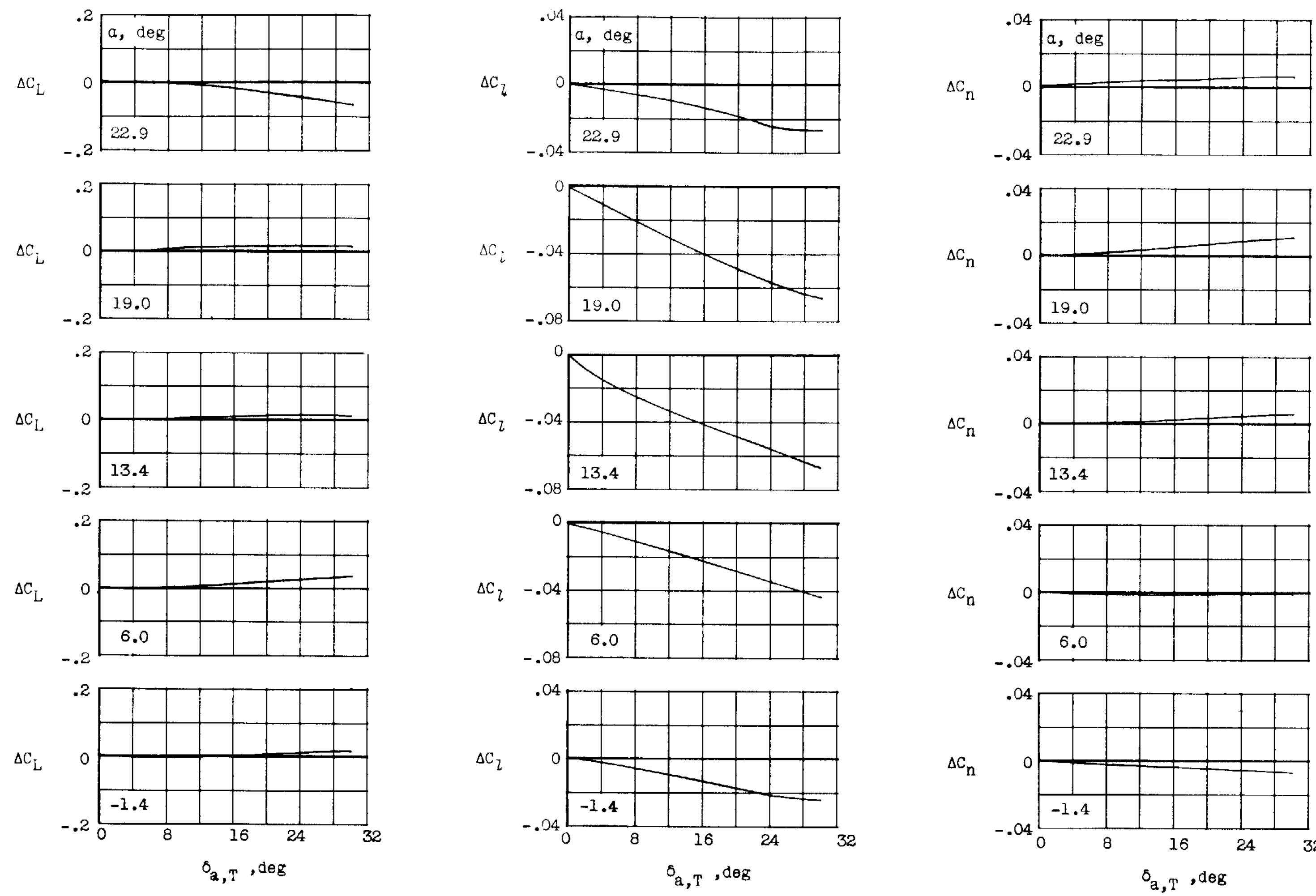
(a) $\delta_f = 47^o$; $\delta_{a,R} = 0^o$; $\delta_n = 50^o$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 15.- Effect of aileron deflection on the values of C_L, C_l, and C_n for either half- or full-span flap configurations. $i_t = 0^o$. With boundary-layer control.



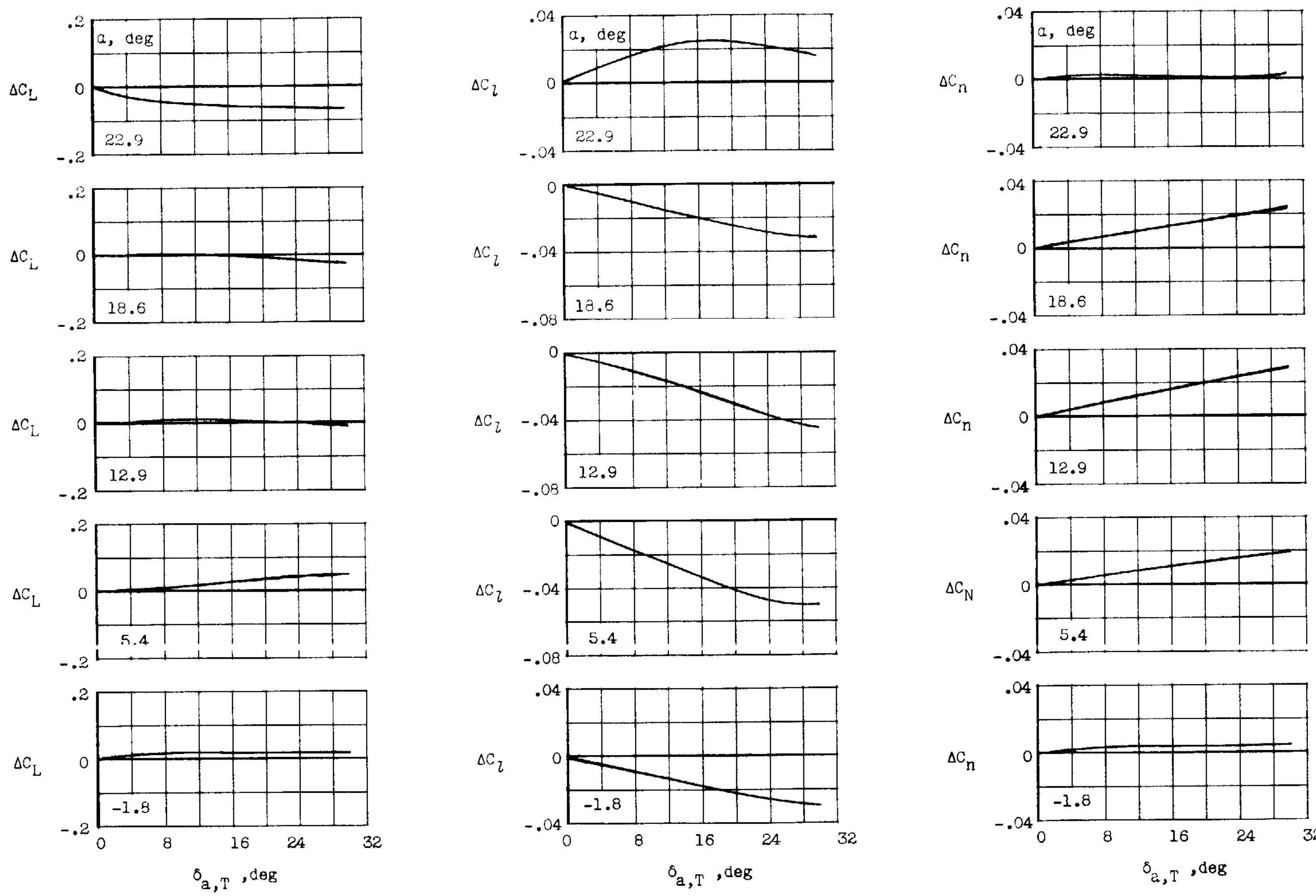
(b) $\delta_f = 47^{\circ}$; $\delta_{a,R} = 47^{\circ}$; $\delta_n = 50^{\circ}$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 15.- Concluded.



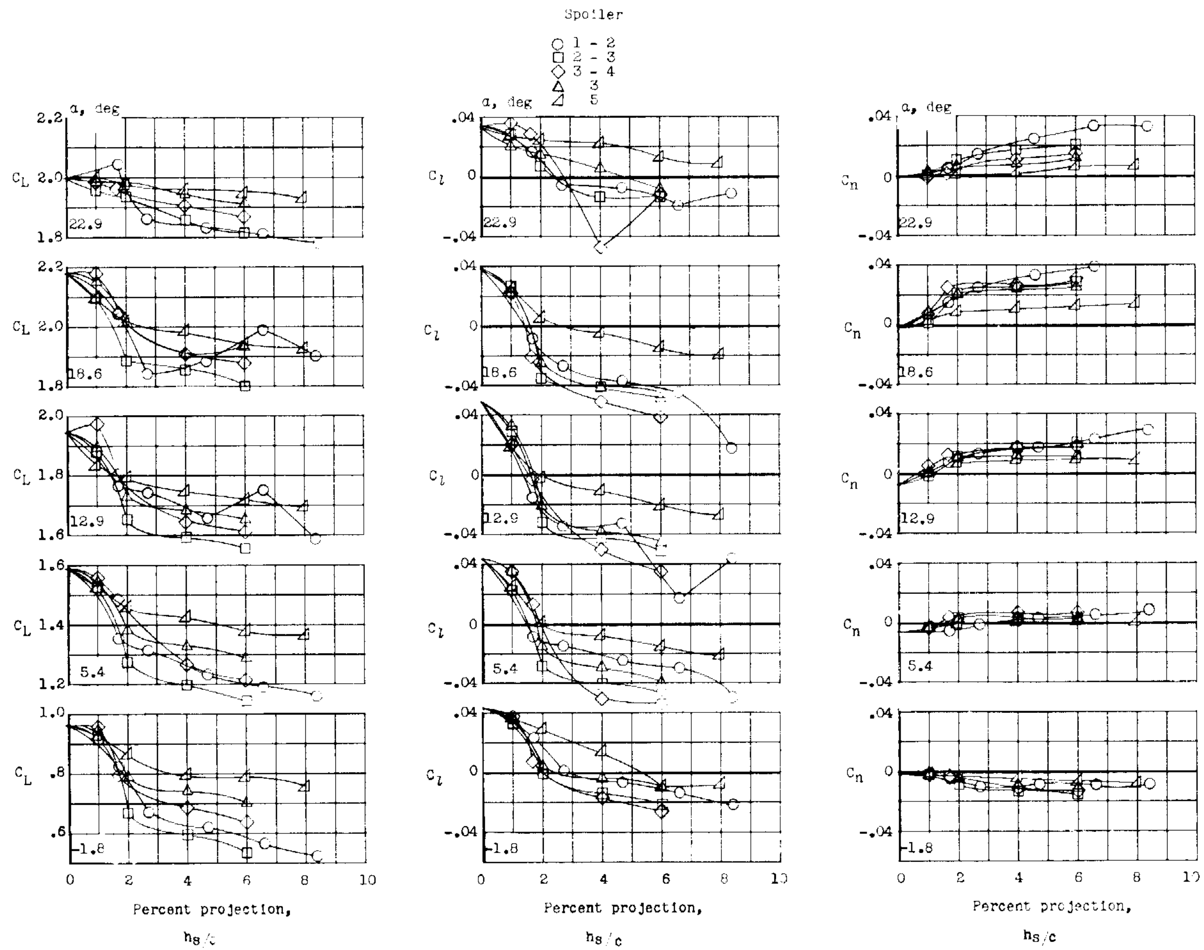
(a) $\delta_f = 47^\circ$; $\delta_{a(\text{neutral pos.})} = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 16.- Incremental values of C_L, C_l, and C_n resulting from differentially deflecting the left-hand aileron up and the right-hand aileron down at a ratio (up to down) of 1 to 2. $i_t = 0^\circ$. With boundary-layer control.



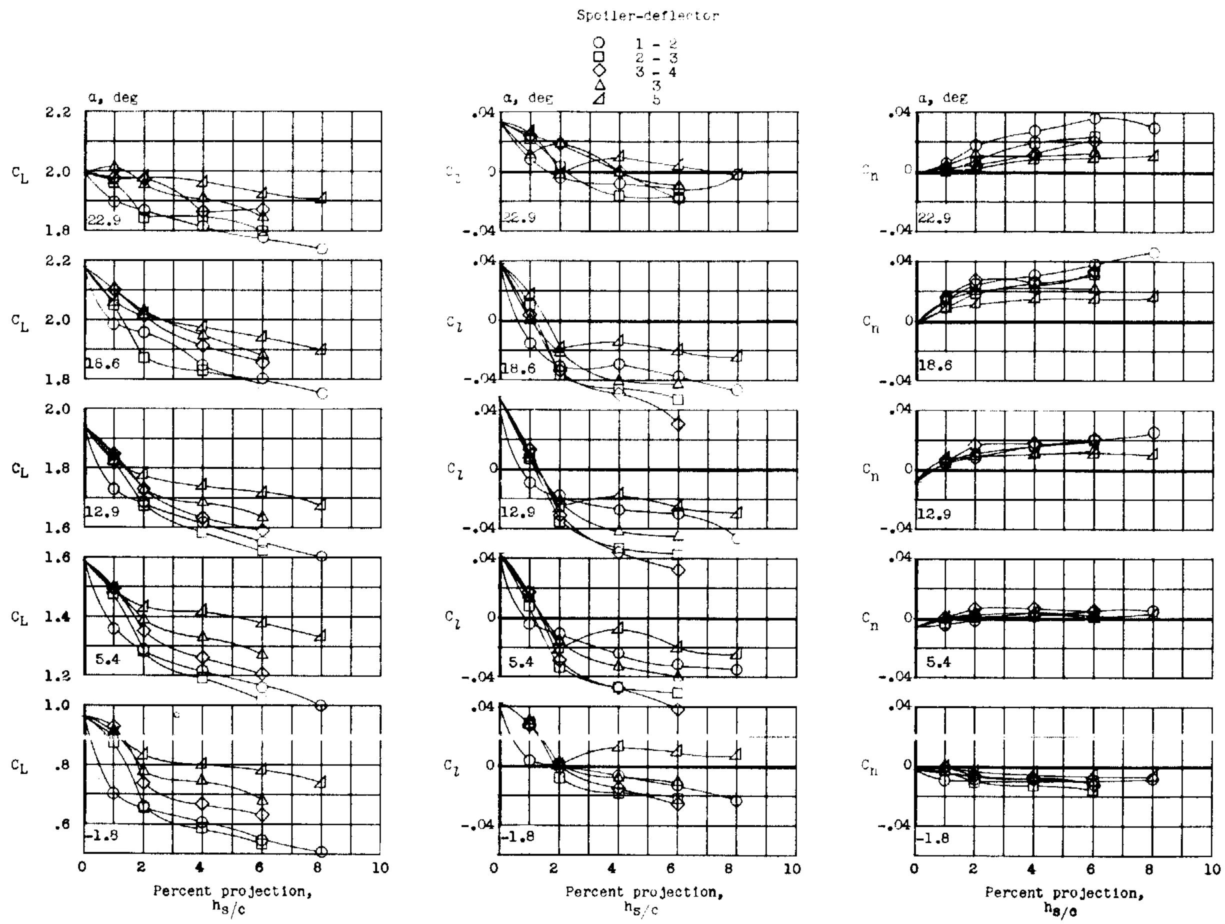
(b) $\delta_f = 47^\circ$; $\delta_{a(\text{neutral pos.})} = 30^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 16.- Concluded.



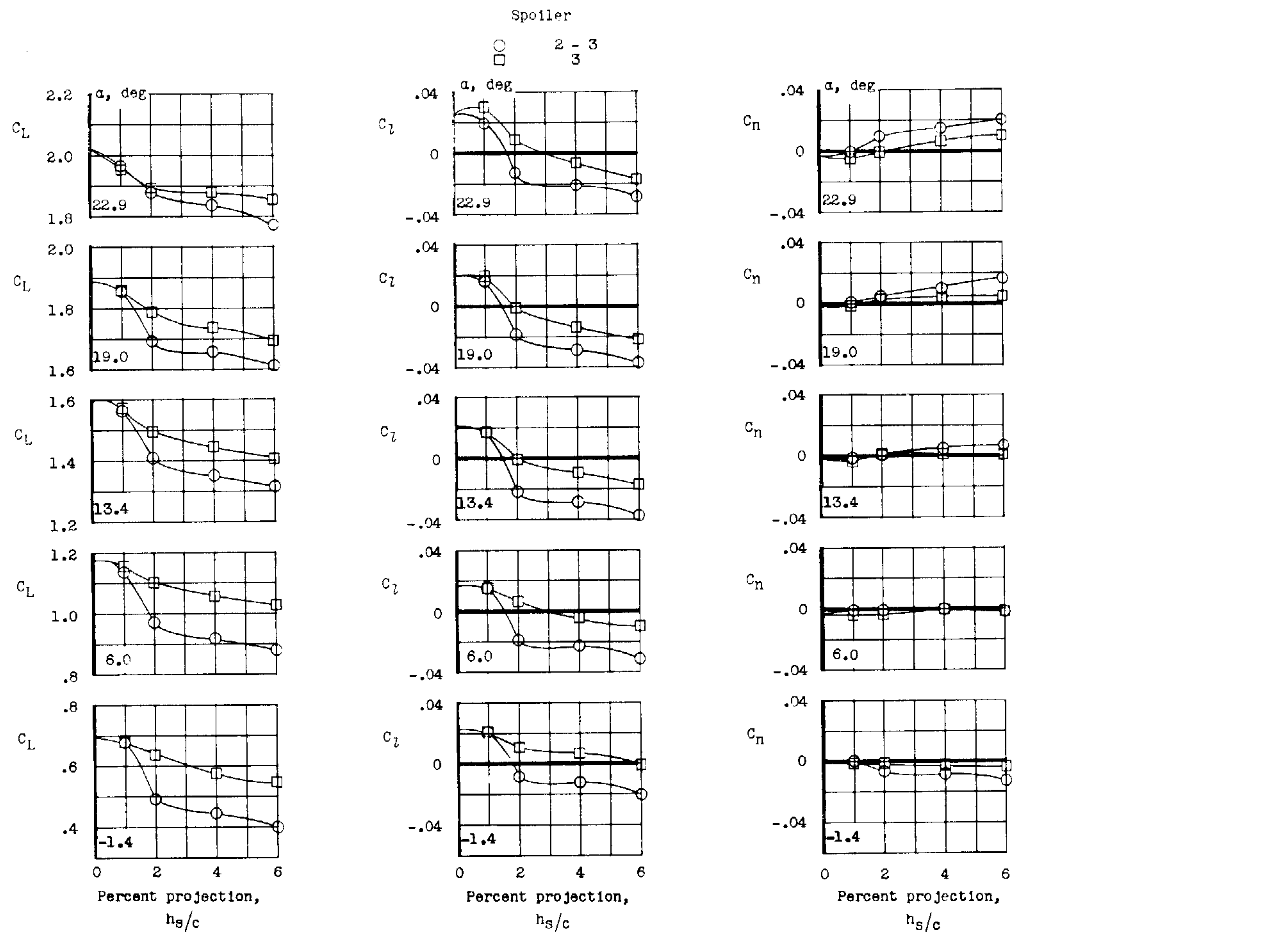
(a) Spoiler only. $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 17.- Effect of spoiler and spoiler-deflector deflection on the values of C_L, C_l, and C_n for the full-span flap configuration. $i_t = 0^\circ$. With boundary-layer control.



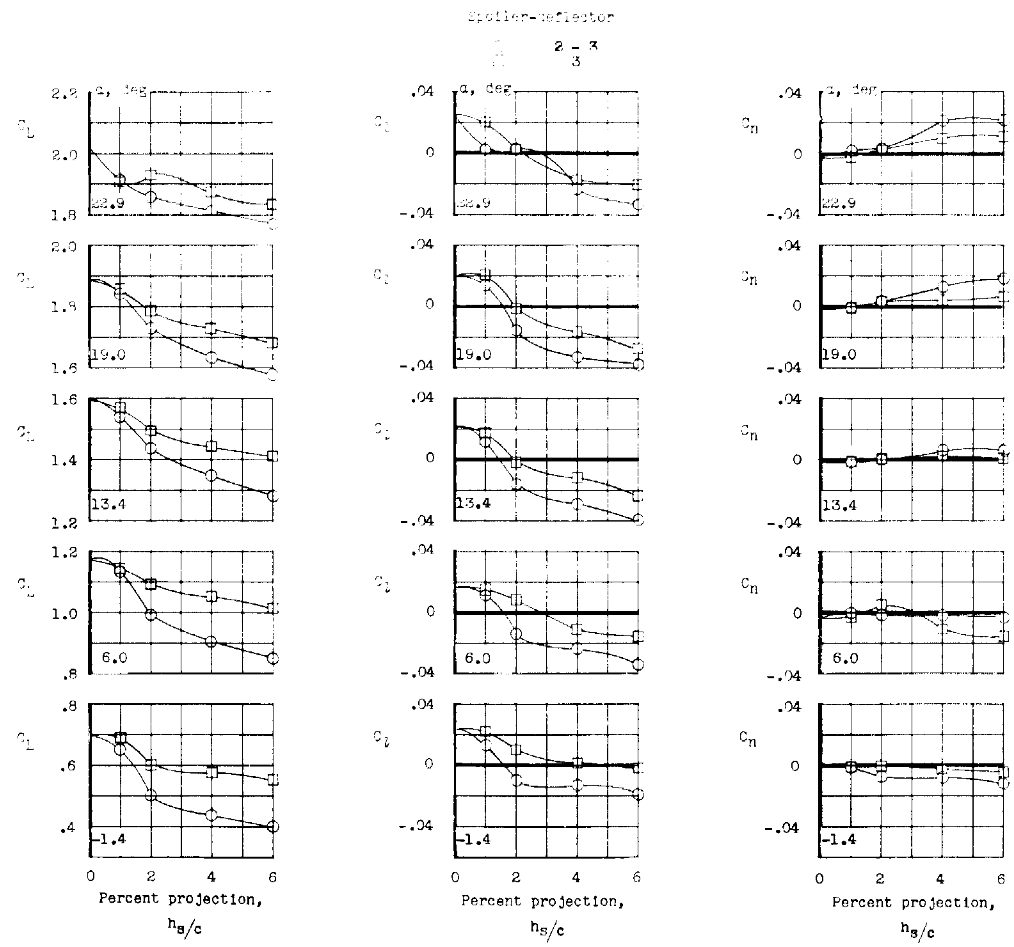
(b) Spoiler-deflector combination; 2 to 1 ratio. $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 17.- Concluded.



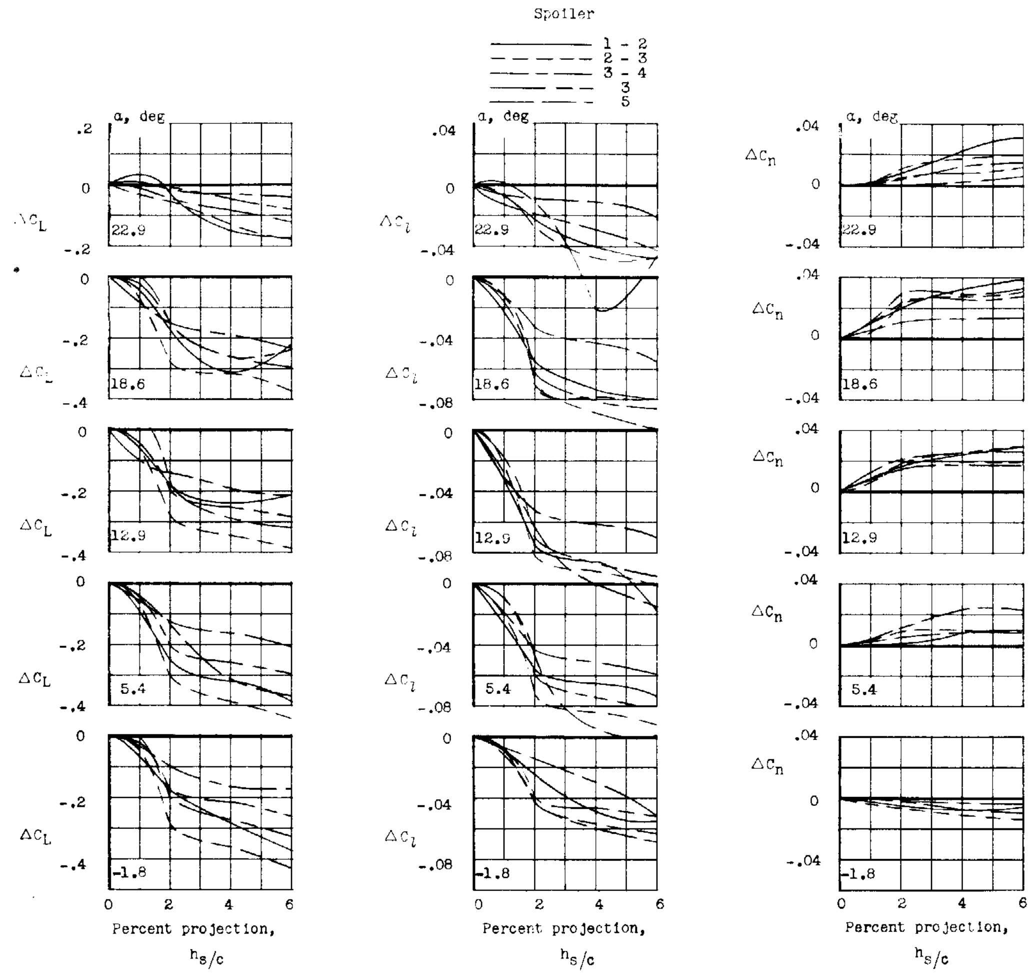
(a) Spoiler only. $\delta_f = 47^\circ$; $\delta_a = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 18.- Effect of spoiler and spoiler-deflector deflection on the values of C_L, C_l, and C_n for the half-span flap configuration. $i_t = 0^\circ$. With boundary-layer control.



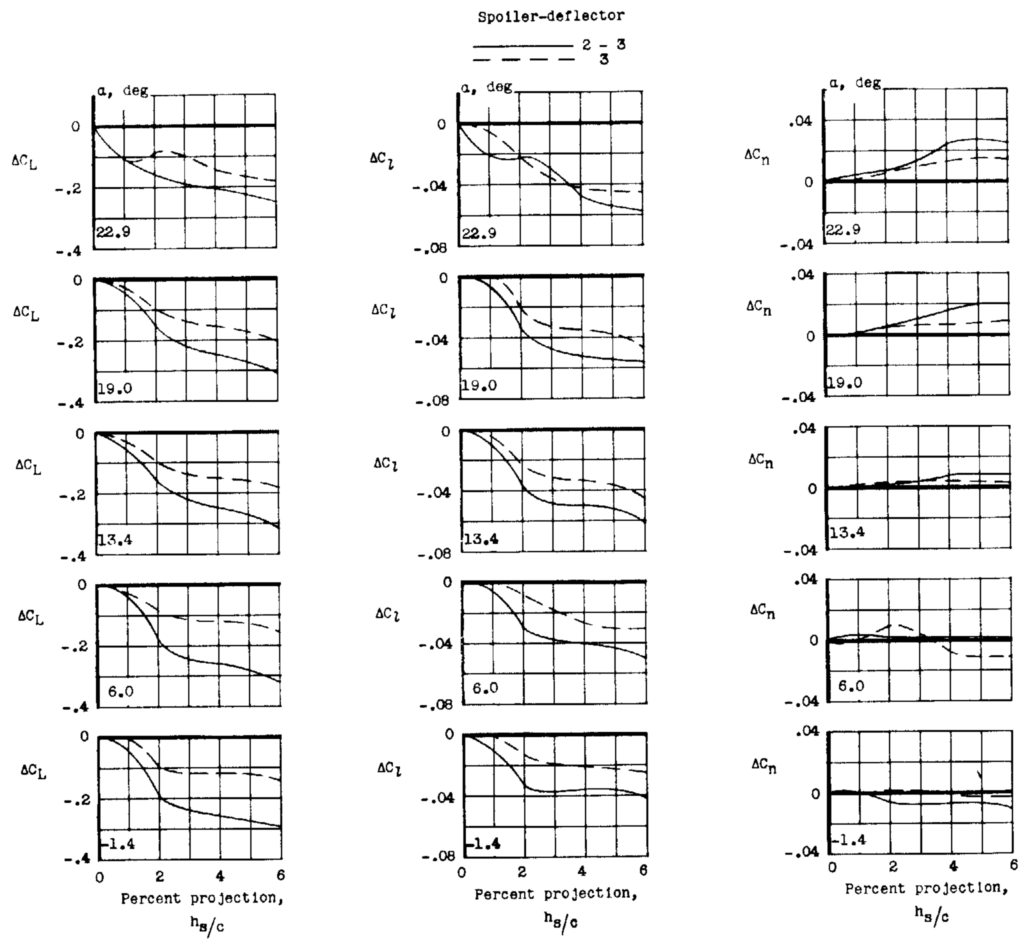
(b) Spoiler-deflector combination; 2 to 1 ratio. $\delta_f = 47^\circ$; $\delta_a = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 18.- Concluded.



(a) Spoiler only. $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 19.- Incremental values of C_L, C_l, and C_n resulting from spoiler and spoiler-deflector deflection. Full-span flap configuration. $i_t = 0^\circ$. With boundary-layer control.



(b) Spoiler-deflector combination; 2 to 1 ratio. $\delta_f = 47^{\circ}$; $\delta_a = 0^{\circ}$; $\delta_n = 50^{\circ}$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

Figure 20.- Concluded.

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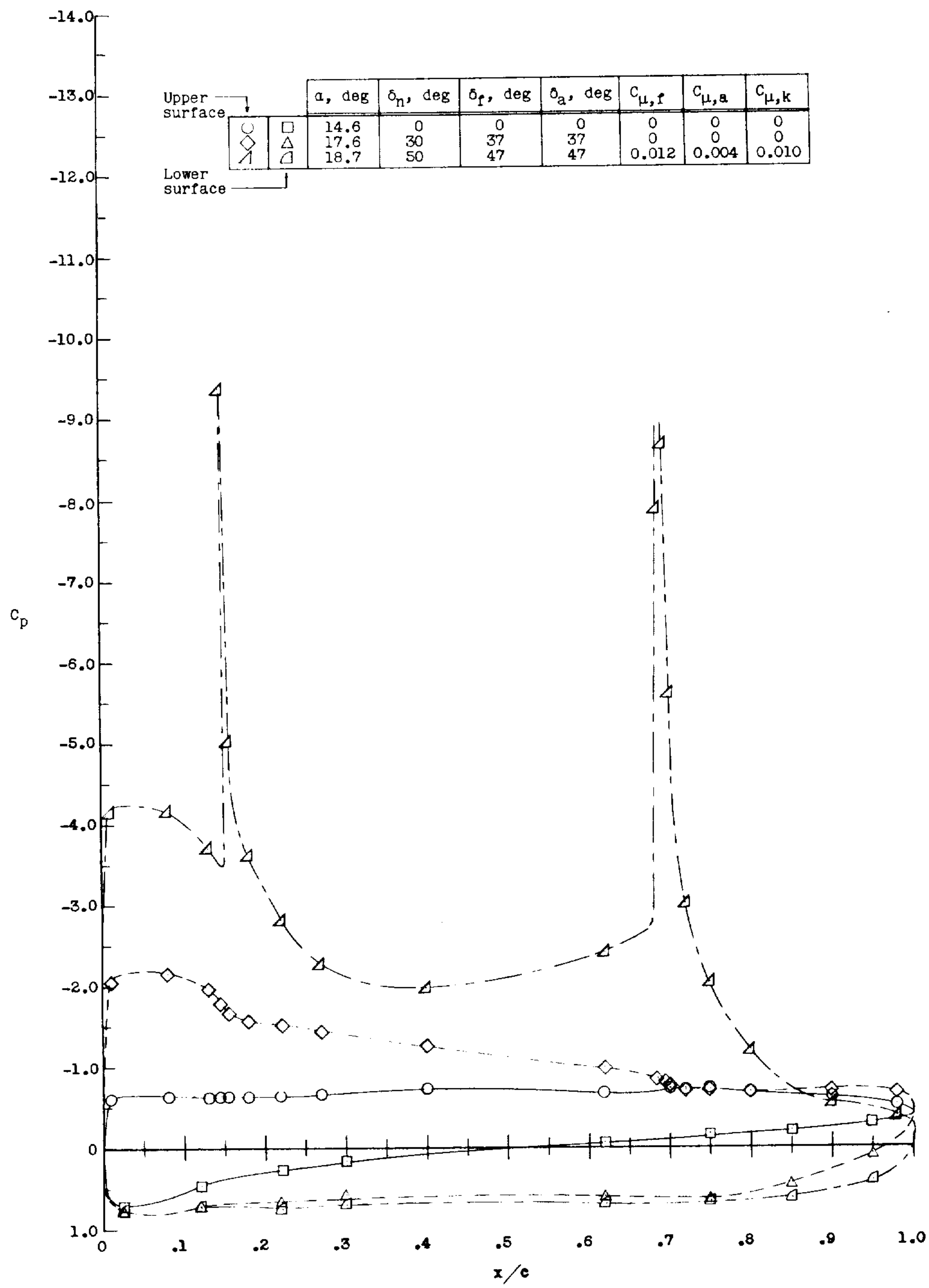
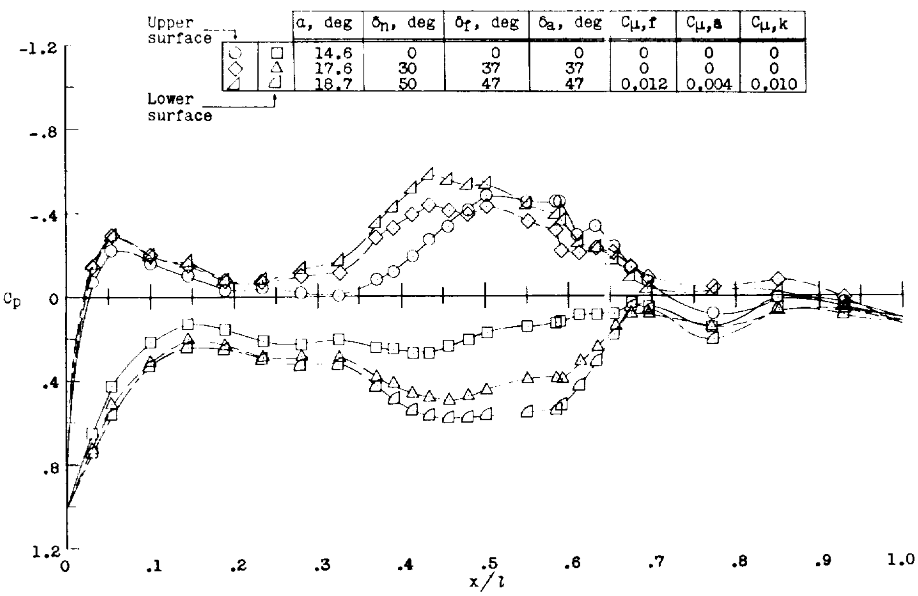
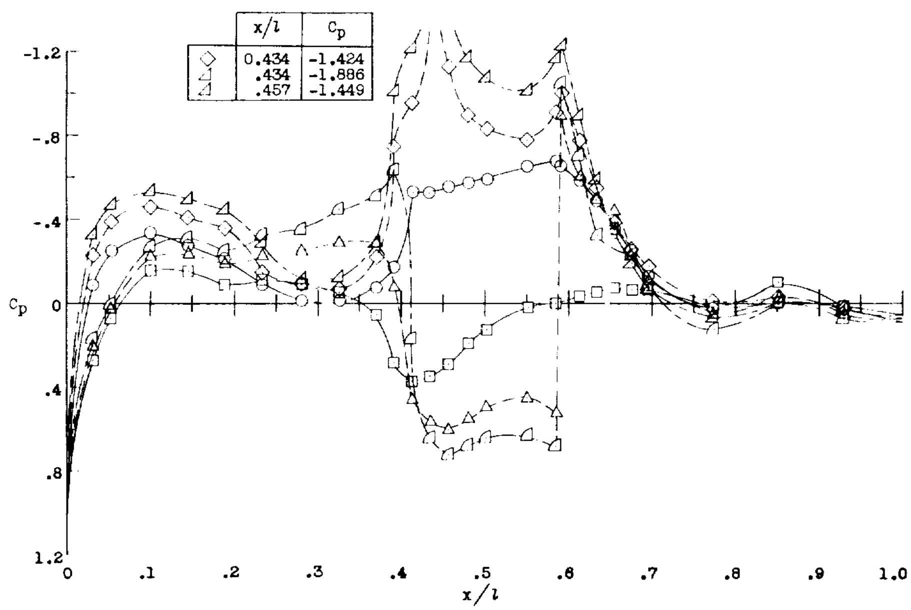


Figure 21.- Typical wing chordwise pressure distribution at station 6 with and without boundary-layer control applied. (Angle of attack near maximum lift.)



(a) Station 1.



(b) Station 2.

Figure 22.- Typical fuselage chordwise pressure distribution at stations 1 and 2 with and without boundary-layer control applied. (Angle of attack near maximum lift.)

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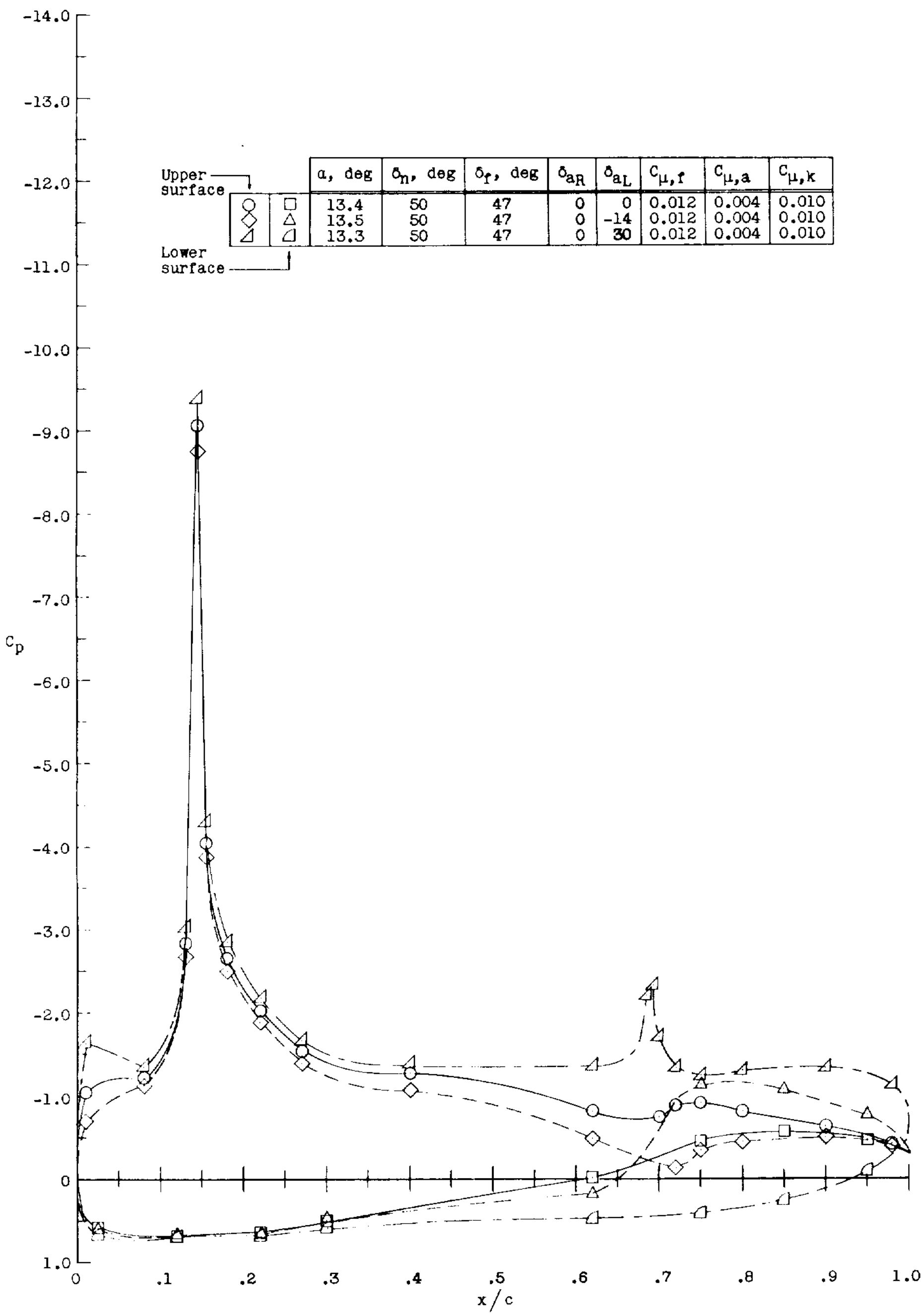
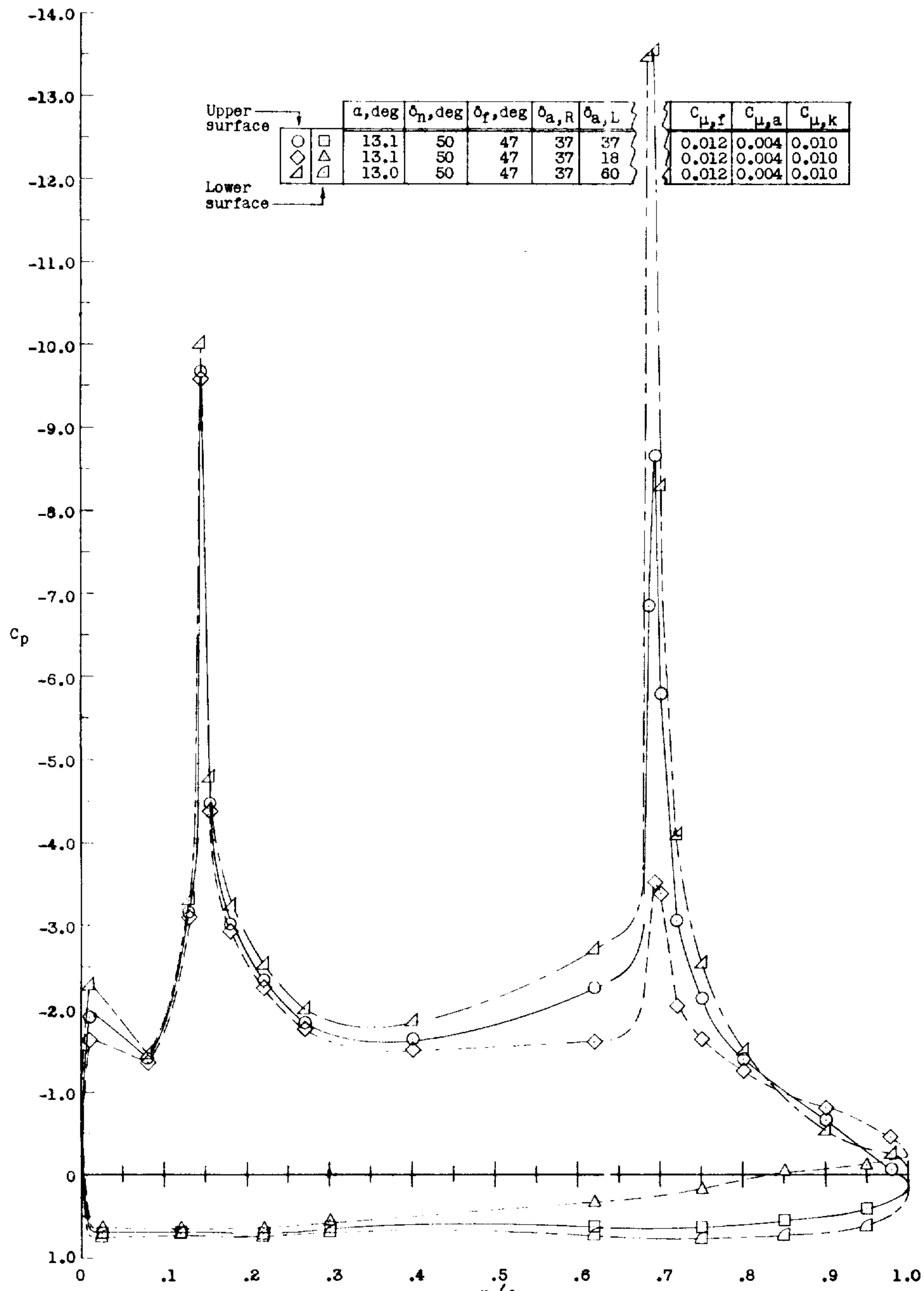


Upper surface	Lower surface	α, deg	δ_n, deg	δ_f, deg	δ_{a_R}	δ_{a_L}	$C_{\mu,f}$	$C_{\mu,a}$	$C_{\mu,k}$
○	□	13.4	50	47	0	0	0.012	0.004	0.010
◇	△	13.5	50	47	0	-14	0.012	0.004	0.010
◺	◺	13.3	50	47	0	30	0.012	0.004	0.010

(a) Half-span flap.

Figure 23.- Effect of aileron deflection on the chordwise pressure distribution at station 6 for the half- and full-span flap configuration.



(b) Full-span flap.

Figure 23.- Concluded.

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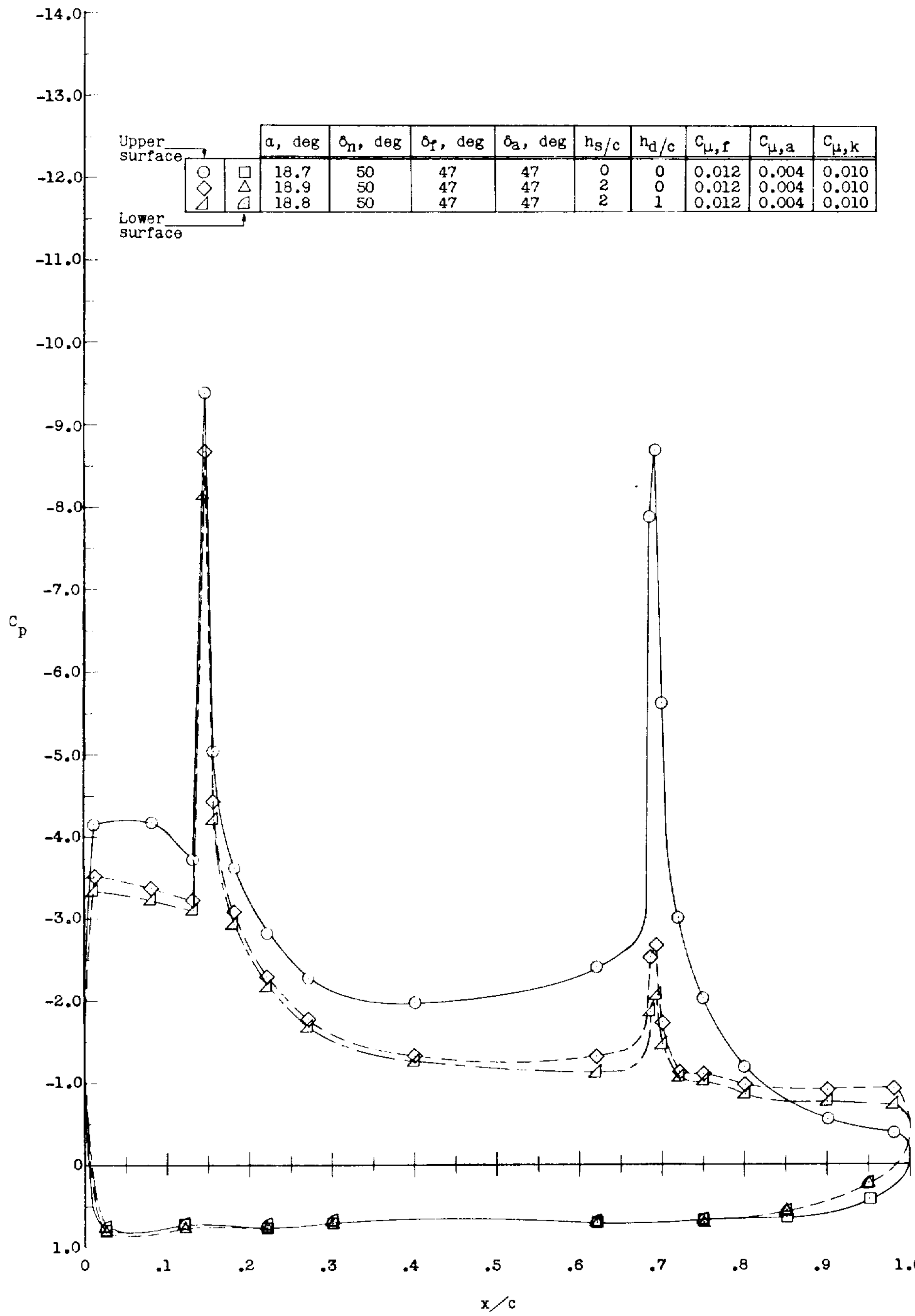
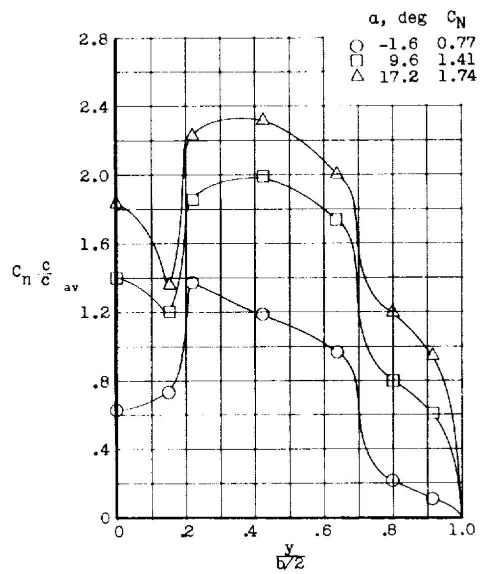
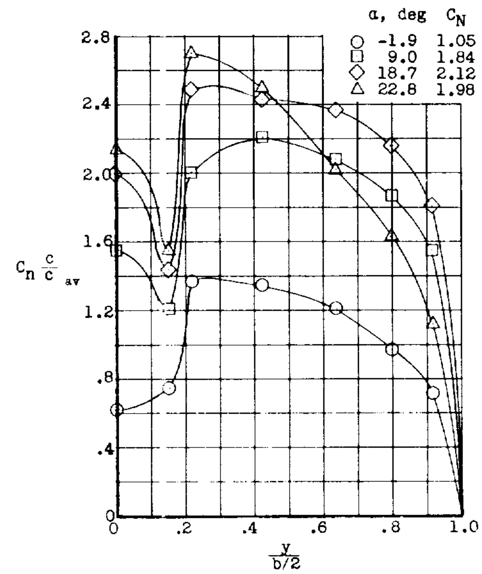


Upper surface	Lower surface	α, deg	δ_n, deg	δ_f, deg	δ_a, deg	h_s/c	h_d/c	$C_{\mu,f}$	$C_{\mu,a}$	$C_{\mu,k}$
○	□	18.7	50	47	47	0	0	0.012	0.004	0.010
◇	△	18.9	50	47	47	2	0	0.012	0.004	0.010
◺	◺	18.8	50	47	47	2	1	0.012	0.004	0.010

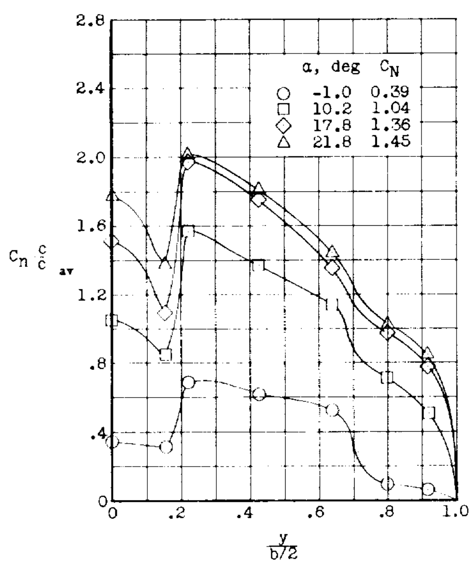
Figure 24.- Effect of projection of spoiler 3 deflection and spoiler-deflector 3 deflection on the chordwise pressure distribution at station 6.

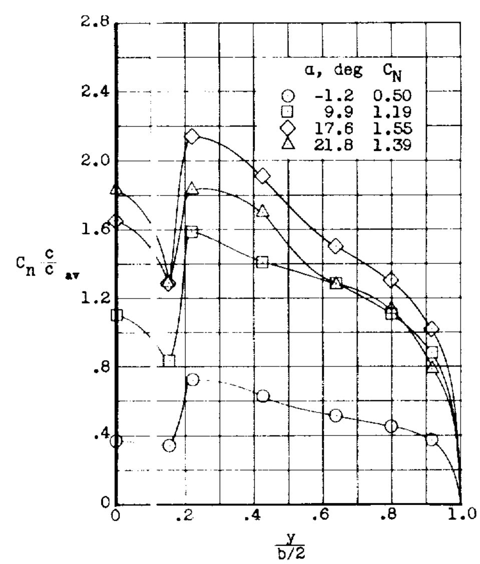




(a) $\delta_f = 47^\circ$; $\delta_a = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.

(b) $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.





(c) $\delta_f = 37^\circ$; $\delta_a = 0^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

(d) $\delta_f = 37^\circ$; $\delta_a = 37^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.

Figure 25.- Span-loading characteristics of several wing configurations.

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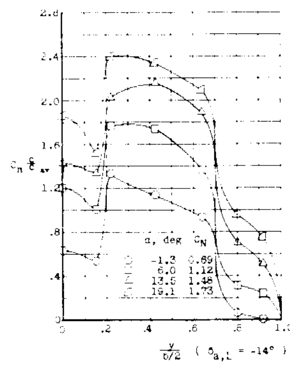
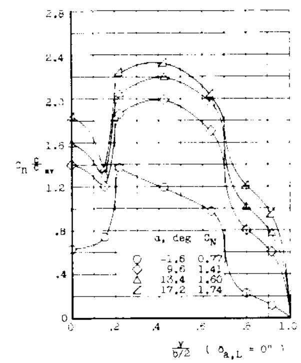
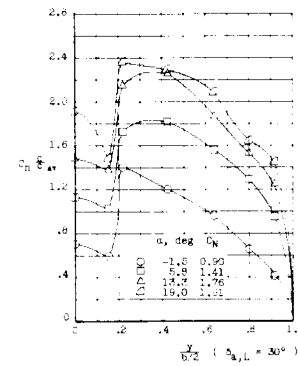


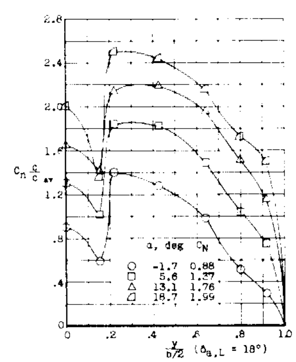
(a) $\delta_f = 47^\circ$; $\delta_{a,R} = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.



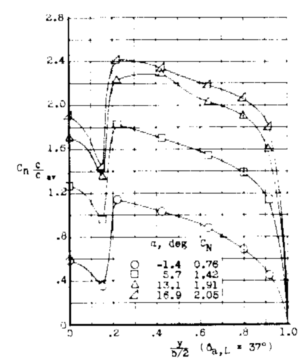
(b) $\delta_f = 47^\circ$; $\delta_{a,R} = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.



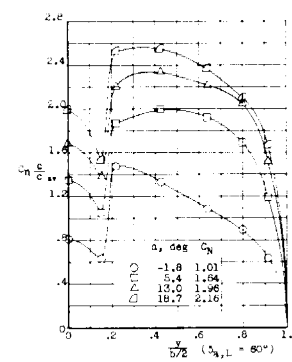
(c) $\delta_f = 47^\circ$; $\delta_{a,R} = 0^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0.010$.



(d) $\delta_f = 47^\circ$; $\delta_{a,R} = 37^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.



(e) $\delta_f = 47^\circ$; $\delta_{a,R} = 37^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.



(f) $\delta_f = 47^\circ$; $\delta_{a,R} = 37^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$.

Figure 26.- Span-loading characteristics of several wing configurations with and without blowing over the ailerons.

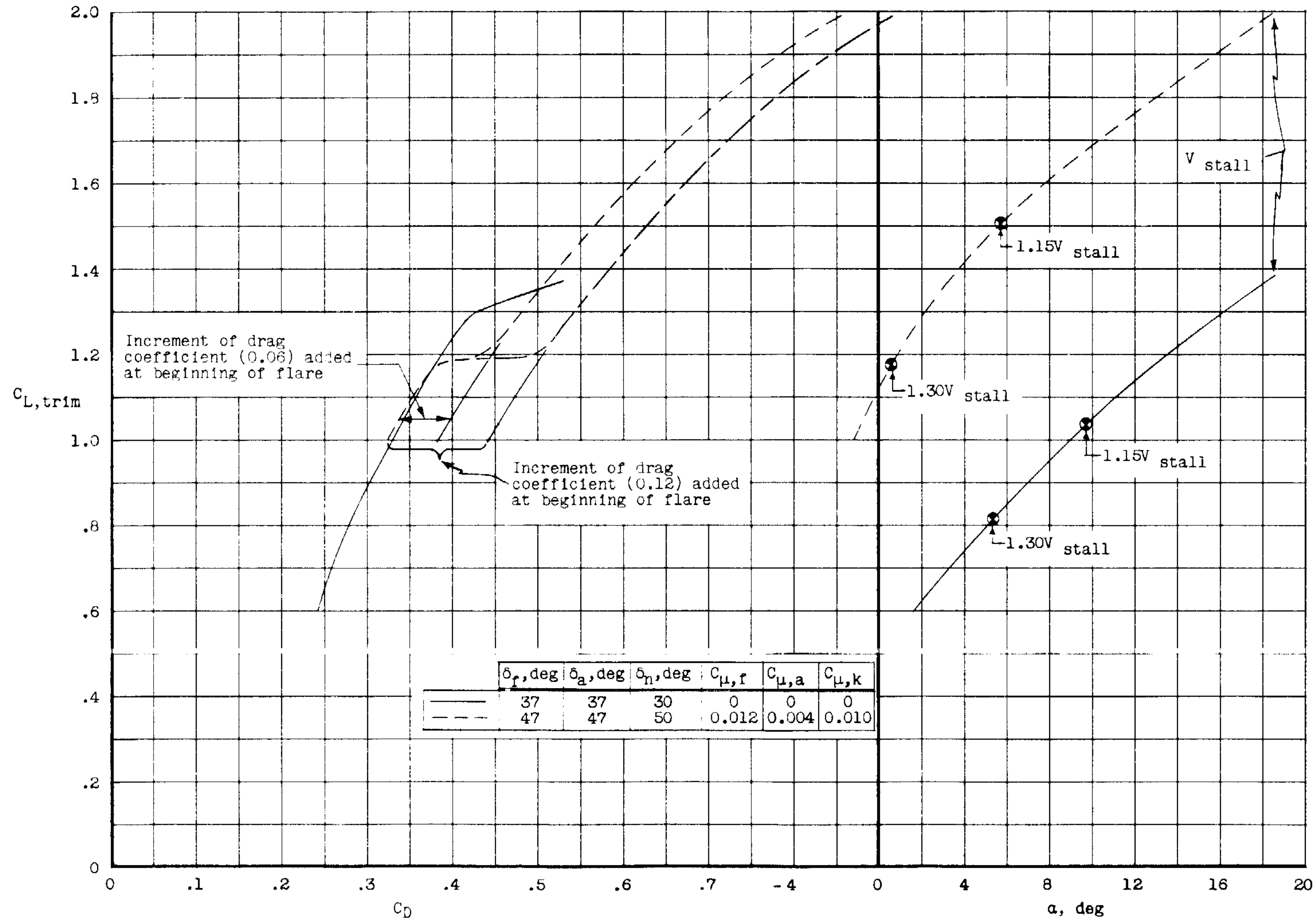
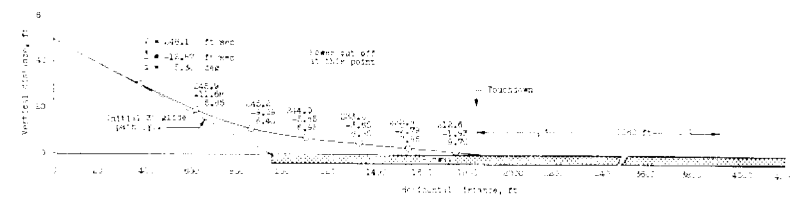
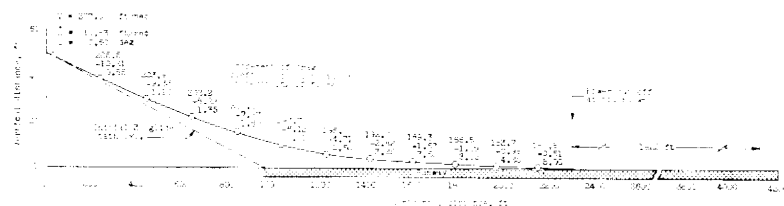


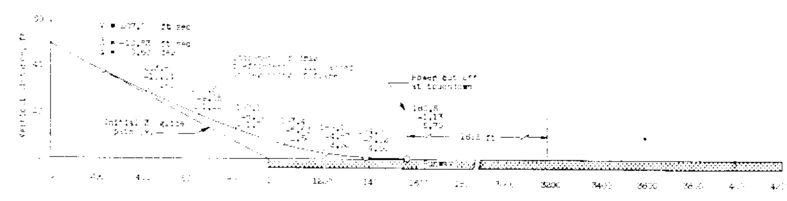
Figure 27.- Trim lift and drag characteristics of two landing configurations used for landing flare analysis. $z/\bar{c} = -0.09$. $W/S = 60$.



(a) Airplane configuration without boundary-layer control; $\delta_f = 37^\circ$; $\delta_a = 37^\circ$; $\delta_n = 30^\circ$; $C_{\mu,f} = 0$; $C_{\mu,a} = 0$; $C_{\mu,k} = 0$.



(b) Airplane configuration with boundary-layer control; $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$ (at initiation of flare).



(c) Airplane configuration with boundary-layer control; $\delta_f = 47^\circ$; $\delta_a = 47^\circ$; $\delta_n = 50^\circ$; $C_{\mu,f} = 0.012$; $C_{\mu,a} = 0.004$; $C_{\mu,k} = 0.010$ (at initiation of flare).

Figure 28.- Landing flare over 50-foot obstacle and ground-roll distance of airplane configurations with and without boundary-layer control.

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